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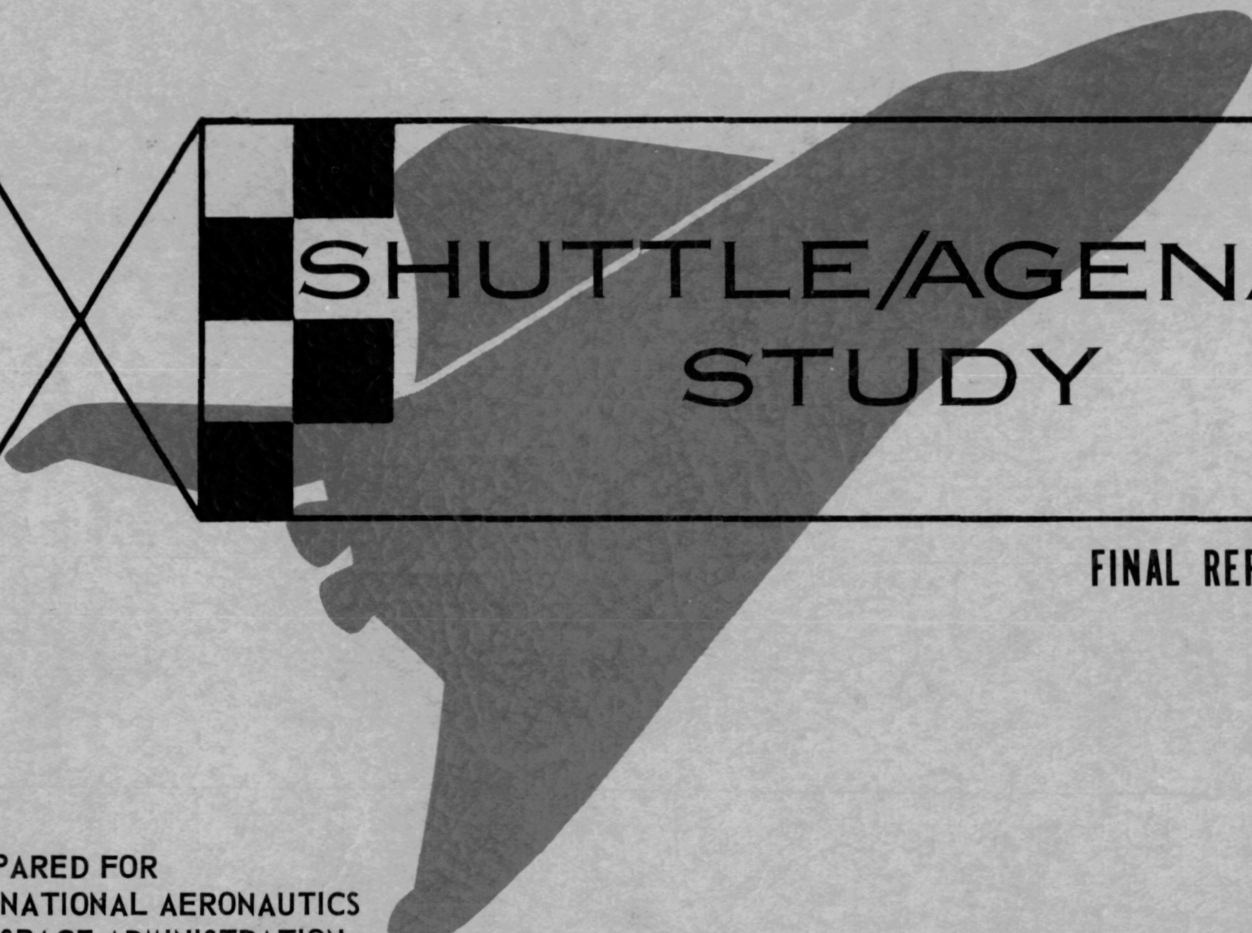

SHUTTLE/AGENA STUDY

FINAL REPORT

PREPARED FOR
THE NATIONAL AERONAUTICS
AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER • HOUSTON, TEXAS

CONTRACT NAS9-11949

LMSC
SPACE SYSTEMS
DIVISION

ANNEX A

ASCENT AGENA CONFIGURATION

LOCKHEED MISSILES & SPACE COMPANY, INC.
A SUBSIDIARY OF LOCKHEED AIRCRAFT CORPORATION
SUNNYVALE, CALIFORNIA

25 February 1972

LMSC-D152635
Series I

SHUTTLE/AGENA STUDY FINAL REPORT

Annex A
ASCENT AGENA CONFIGURATION

Submitted to the
National Aeronautics and Space Administration
Manned Spacecraft Center
Houston, Texas

LOCKHEED MISSILES & SPACE COMPANY, INC.
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FOREWORD

Annex A contains supplemental data pertinent to the Shuttle/Agena Study Final Report. Although not required under Contract NAS9-11949, this background information on the Ascent Agena should prove helpful in any evaluation of Agena capabilities for the proposed space tug missions. It is therefore included as a useful adjunct to the Final Report.

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INTRODUCTION

Section 1
INTRODUCTION

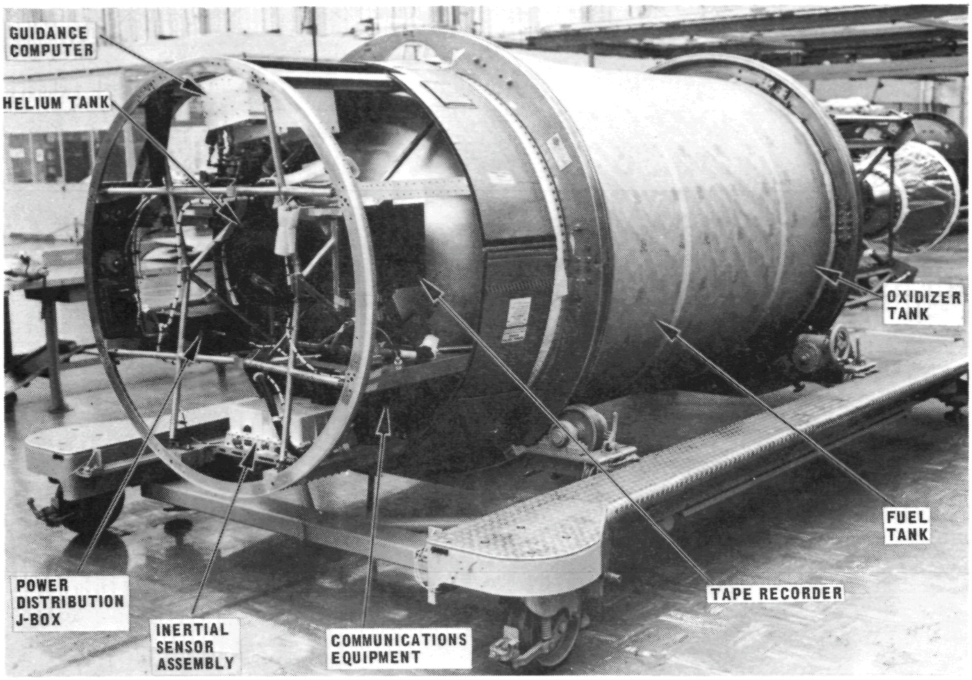
The Ascent Agena was developed for use as one of the upper stages on a multistage ascent vehicle. It is adaptable to many combinations of booster, payload, and launch site or downrange support hardware. Its mission capabilities include onboard functional programming of flight events by computer; single-, dual-, or triple-start engine burn sequence; all-inertial attitude sensing; ground command response; data recording and telemetry; and full provisions for spacecraft or other payload protection and support.

The range of Ascent Agena capabilities to be employed in a particular space launch mission depends on the ultimate hardware configuration. It may be used as the second or subsequent stage in any multistage vehicle with which it is compatible; typically, with the Thorad, Atlas, or Titan families of lower-stage vehicles. It may be programmed to achieve a specific orbital pattern, to assume and maintain a sequence of desired attitudes, and to initiate subsequent stage or spacecraft functions. These capabilities have been demonstrated repeatedly for various types of missions, ranging from low through synchronous earth orbits and earth-escape trajectories.

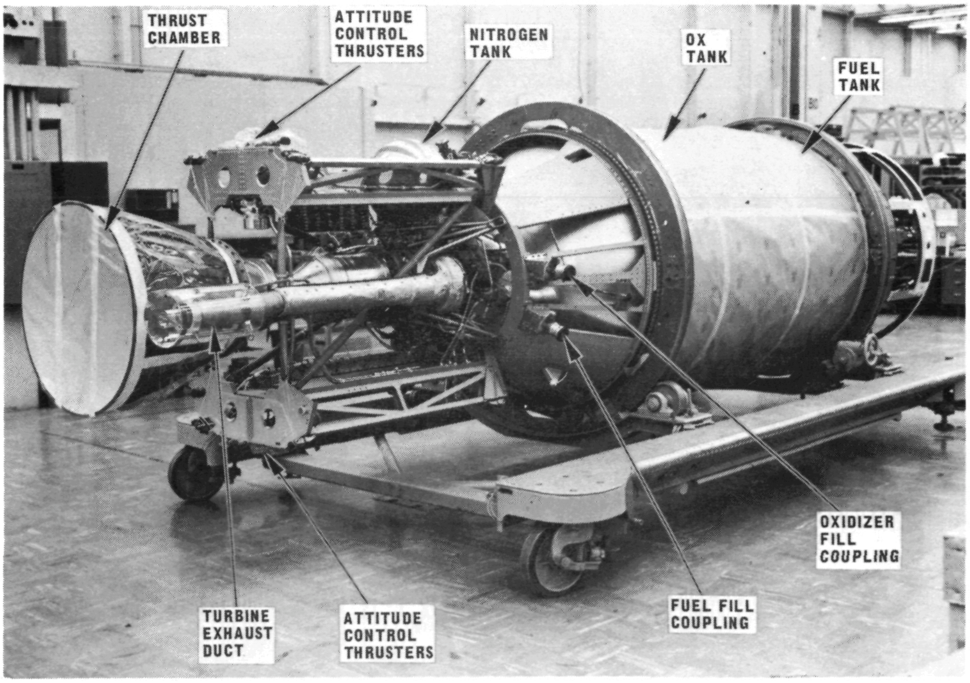
1.1 DESIGN CONCEPT

The basic design concept for the Ascent Agena is to optimize reliability, flexibility, performance capabilities, and economy through the use of tested and flight-proven hardware. Agena hardware designations—basic, optional, or mission-peculiar—reflect the extent to which implementations of this concept have been carried out.

The basic vehicle (Fig. 1-1) consists of structural and operating equipment common to the requirements of most programs. Provisions are incorporated in the basic vehicle for installation of optional kits to add specific features and capabilities. Through the



Forward View



Aft View

Fig. 1-1 Ascent Agena

process of equipment selection, the program can acquire a vehicle with desirable features without incurring the weight penalties and economic disadvantages that might be expected from an all-purpose vehicle.

General characteristics of the Ascent Agena are tabulated below. In the pages that follow, a brief genealogy of the Agena vehicle gives insight into the evolution of this highly successful space vehicle and documents its solid record of achievement.

Propellants	IRFNA/UDMH
Vehicle Diameter (in.)	60
Vehicle Length (in.)	248.5
Nominal Propellant Gross Weight (lb)	13,400
Nominal Mixture Ratio	2.57
Nominal Specific Impulse (sec)	291
Thrust (lb)	16,000
Start Capability	1, 2, or 3
Total Engine Burn Time (sec)	240
Dry Weight (lb)	1225
Guidance System	IGS
Engine	BAC 8096

1.2 AGENA DEVELOPMENT HISTORY

In more than 320 flights, the Agena has been a reliable, cost-effective upper stage, acting as an essential element of the launch vehicle. In most of these missions, the Agena has also become an on-orbit satellite, performing broad assignments; on other missions, the Agena has injected separable spacecraft into precise orbits or into trajectories to the moon and planets. Since early 1967, the Agena has achieved a 100-percent success record. The vehicle that compiled this record has undergone many refinements through the years. This evolutionary process is summarized below.

1.2.1 Agena A

In 1956, Lockheed began development and production of the Agena space vehicle. Planners envisioned the Agena as an upper stage vehicle that would supply the added boost needed to send large payloads into space and would serve as a stabilized platform on orbit. The Agena power plant was derived from the existing B-58 Hustler rocket engine. For the first Agena flight in February 1959 on a Thor booster (Figs. 1-2 and 1-3), the engine used JP-4 jet fuel and IRFNA oxidizer and had a burn capability of 100 seconds. Following this flight, the fuel was changed to UDMH, which is hypergolic with IRFNA, and the burn time was increased to 120 seconds. The Agena A series – 18 vehicles in all – conclusively demonstrated that the orbiting upper stage concept fulfilled all mission expectations.

1.2.2 Agena B

The Agena A's success led to inauguration of long-range Agena development plans, and a succession of more sophisticated and higher-performance Agenas evolved. Agena B, a second-generation vehicle which incorporated many improvements, was first launched in October 1960 (Fig. 1-4). Structurally improved and lighter, the nested propellant tanks were doubled in volume, increasing burn time to 240 seconds and permitting a second burn. A triplet propellant injector and an increased expansion ratio nozzle resulted in higher specific impulse to accompany the advantages provided by the shaped injection trajectory. Agena B success led to the demand for greater performance to carry out the more challenging missions for which new systems and devices were being developed.

1.2.3 Agena D

Agena D (Fig. 1-5) was first launched in June 1962; and improvements in methods, materials, and devices were implemented as each major block of vehicles was manufactured. Standardized payload interfaces, an integrated guidance module, modularized communications, and a linear pyrotechnic device for booster separation were introduced. Modification kits adapted the Agena for short-term missions, long-term missions, low-altitude orbits, high-altitude orbits, rendezvous missions, lunar and deep-space probes, stabilized spacecraft, and ascent missions (Figs. 1-6 and 1-7). Development of an advanced propellant tank sump, in which surface tension controls propellant positioning under low-g conditions, eliminated the need for pre-ignition accelerations.

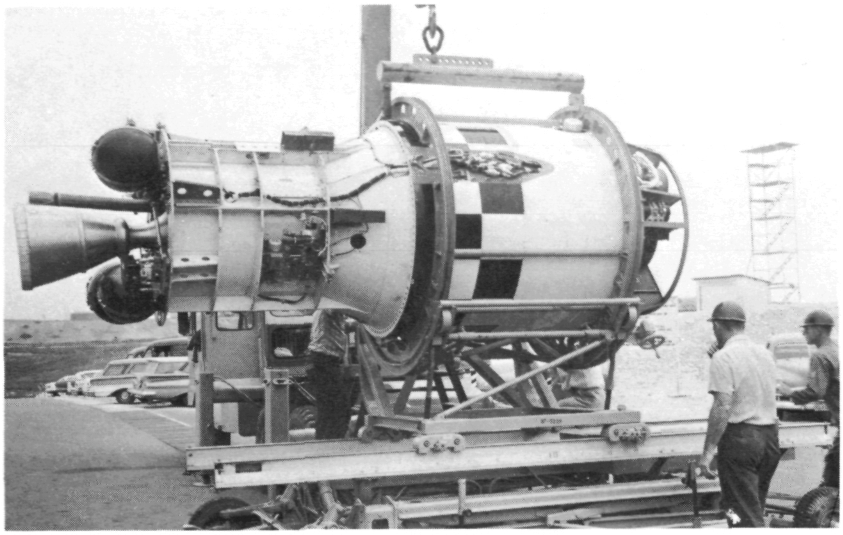


Fig. 1-2 The first Agena A arrives at the new West Coast launch complex in 1959, inaugurating the series of Agenas that have since flown more successful missions than all other upper stages.

Fig. 1-3 Poised on a Thor Booster, Agena A is ready for Launch No. 1.

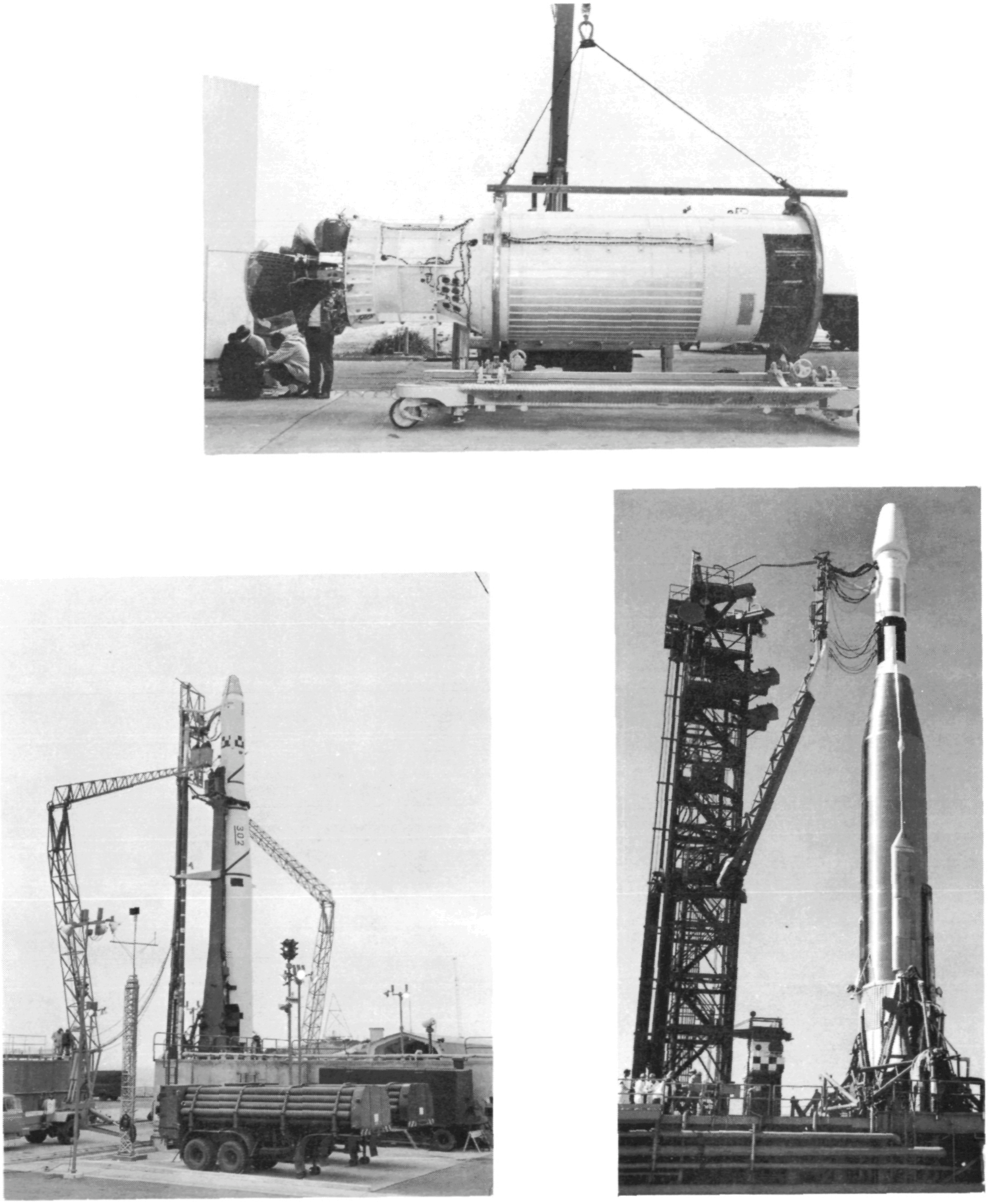


Fig. 1-4 An Agena B is unloaded (top) at WTR in preparation for a 1960 flight. Altogether, 72 Agena B's were flown over a 6-year span. Of these, 44 were boosted by Thor (bottom left); 28 were boosted by Atlas (bottom right).

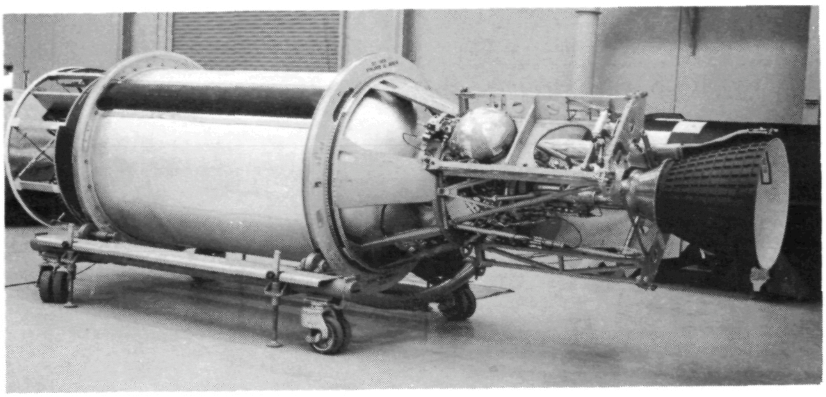


Fig. 1-5 The Agena D was designed for mass production to meet the growing demand in the sixties for a reliable upper stage. The Agena D shown above has satisfactorily completed tests prior to delivery to a using program.

Fig. 1-6 The addition of auxiliary racks to the Agena D forward ring expanded the equipment mounting provisions on spacecraft Agena configurations.

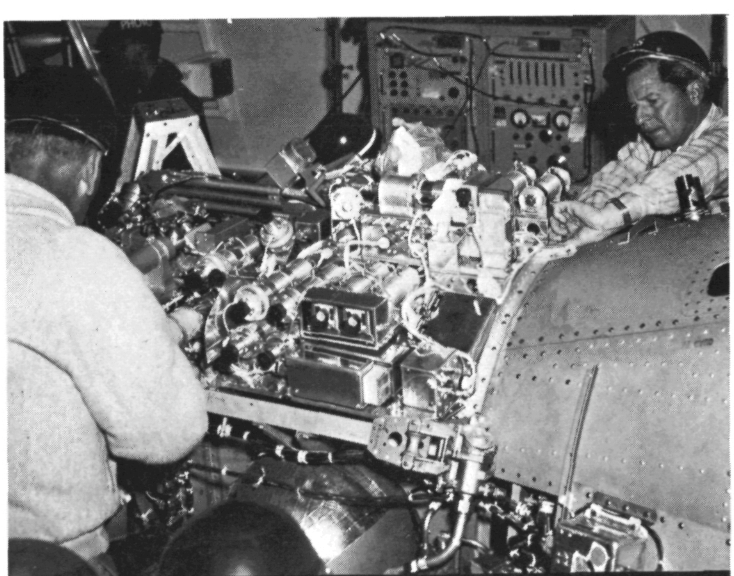


Fig. 1-7 Space was allocated on the Agena D aft rack to permit installation of piggyback experiments, payload support hardware, and solar arrays.

By 1965, both the National Aeronautics and Space Administration and the Air Force were relying on the Agena as an upper stage for launch vehicles of separable spacecraft (Fig. 1-8). Lockheed organized a specialized team to study the unique ascent role, and the conceptual design of an Ascent Agena began in 1967. This vehicle was specifically designed for ascent missions, but proven Agena processes and flight-qualified hardware were used. The special-purpose Agena design concept was presented to ascent vehicle users in 1968; shortly thereafter, the U. S. Air Force contracted for development of the Ascent Agena, which was flown operationally in early 1971.

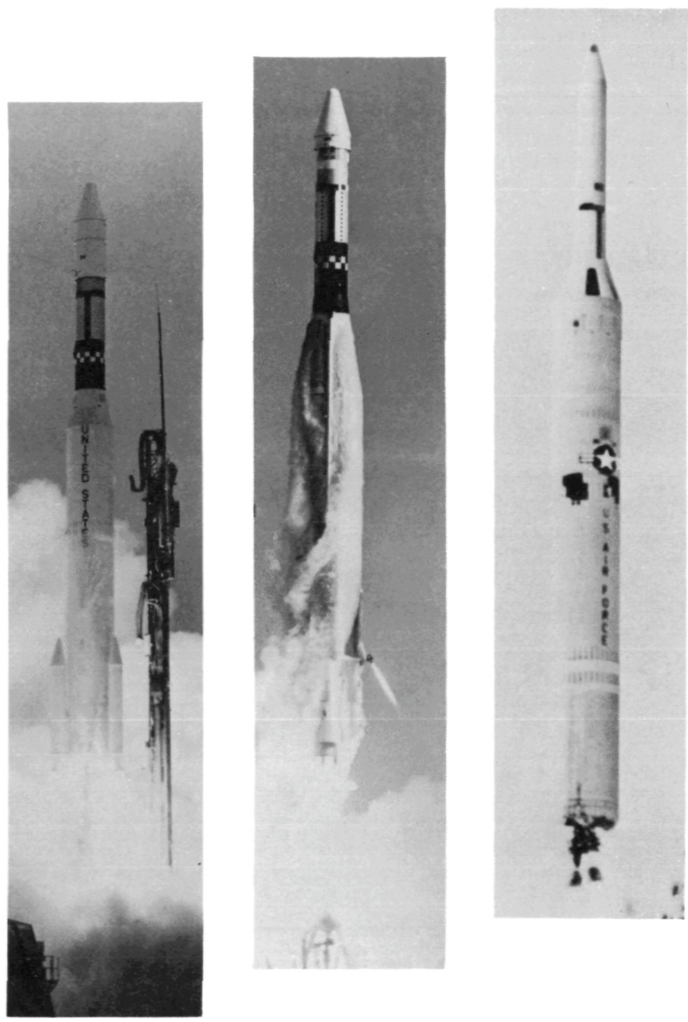


Fig. 1-8 The Agena D is the only vehicle that has flown with all three of the major medium-thrust booster systems. Three of every four Thor and Atlas space booster missions and 60 percent of all unmanned Titan space booster missions have flown with Agena as the upper stage. Flawless performance has resulted in 100-percent success during the past 4 years.

Throughout the years, development of the Agena has kept pace with the rapid advancement of technology and requirements; continued study and development will keep the Agena abreast of future needs. The application of strapdown inertial guidance, lightweight electronics and structures, and simplified design provide an ascent vehicle in keeping with the Agena tradition of dependability, precision, and versatility.

1.2.4 Flight History

A year-by-year breakdown of Agena launches since 1959 is presented in Fig. 1-9.

1.2.5 Agena "Firsts"

In the course of more than 320 flights, the Agena has established many significant technological advances:

- First three-axis stabilized upper stage
- First polar orbiting satellite
- First three-axis-stabilized satellite on orbit
- First nuclear reactor in space
- First in-space engine restart
- First recovery of payloads from space by both air and water retrieval
- First vehicle to provide on-orbit maneuvers
- First to conduct manned orbital rendezvous and space docking (Fig. 1-10)
- First host vehicle for the amateur radio satellite
- First Canadian satellite
- First electric ion engine in orbit
- First joint US/USSR cooperative mission
- First launch of eight satellites with single boost vehicle combination

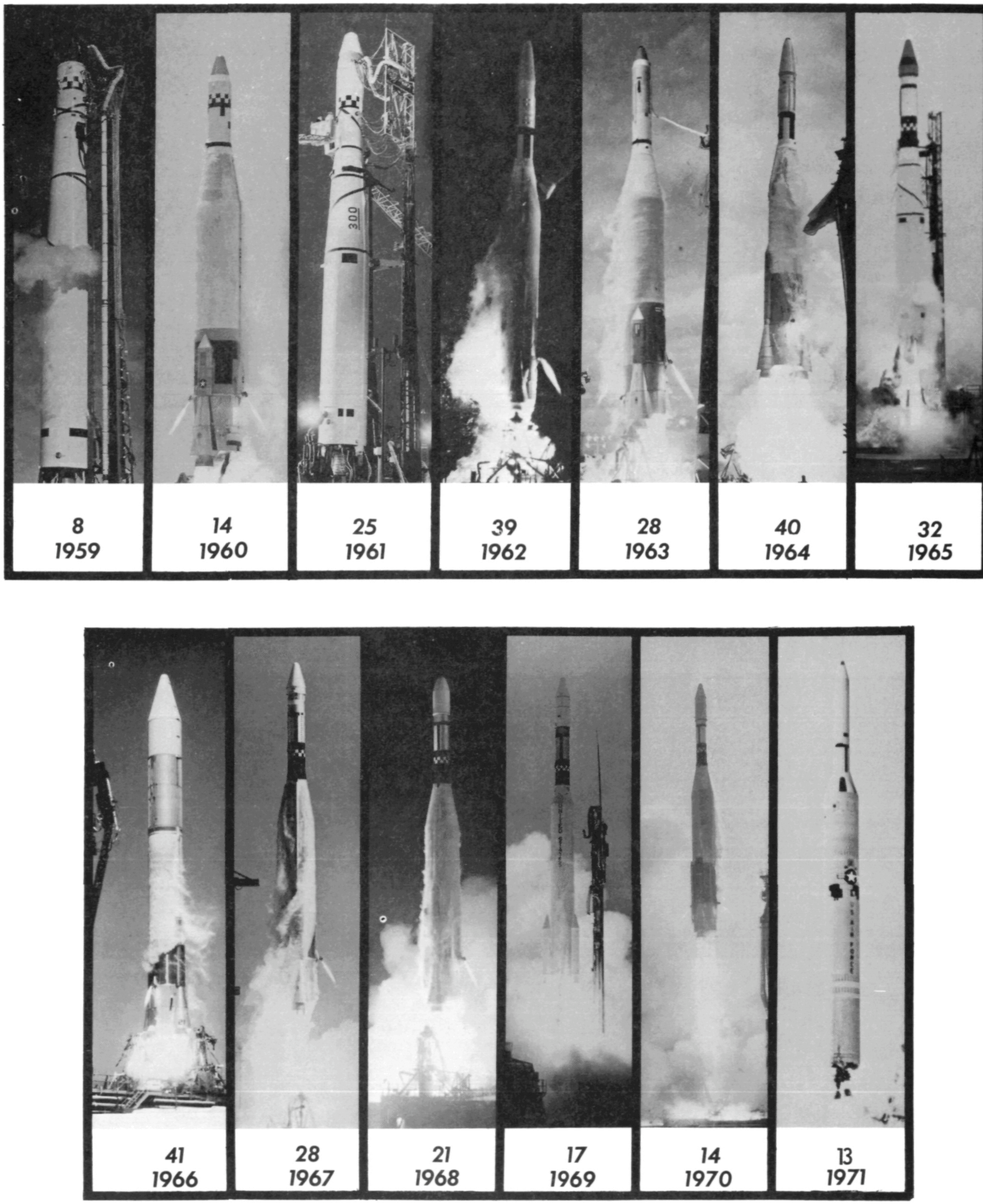


Fig. 1-9 Agena Launches – 1959 through 1971

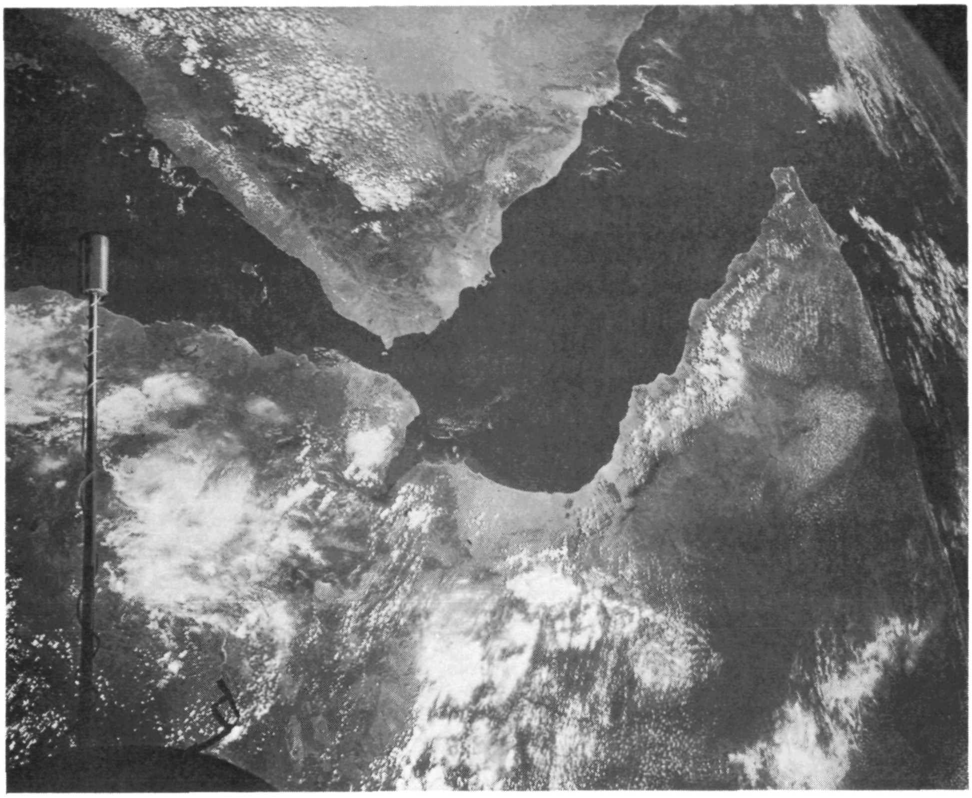


Fig. 1-10 The boom of the Gemini Agena target vehicle is visible to the astronauts as they dock over Ethiopia and Arabia.

1.2.6 Space Applications

In the long history of the Agena from first launch in February 1959 to the present, one of its most outstanding attributes has been versatility. Meeting program requirements ranging from relatively simple orbit injection to complex long-life missions, the Agena has been outfitted with all manner of mission-related equipment, producing many different configurations for flight (Figs. 1-11 through 1-13). This versatility and the availability of proven optional equipment permits Agena adaptation to many future missions. With minor changes in avionics, increased expendables, and greater electrical power provisions, the adaptable Agena becomes a long-life orbiting spacecraft. In this capacity, Agena is ideally suited to provide large volumes of payload mounting space and proven long-term attitude stabilization modes. In all spacecraft missions, the Agena retains the role of ascent vehicle. This dual role is particularly cost effective.

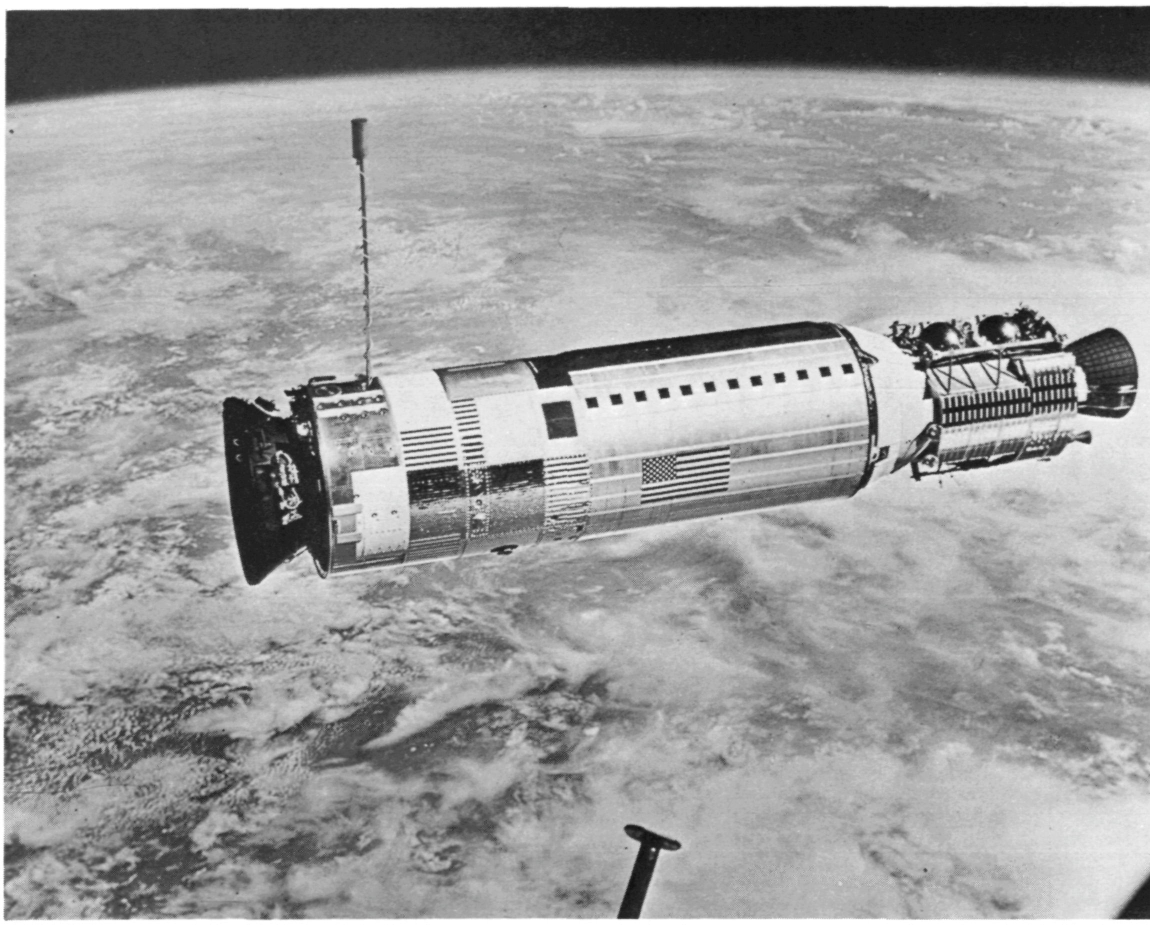
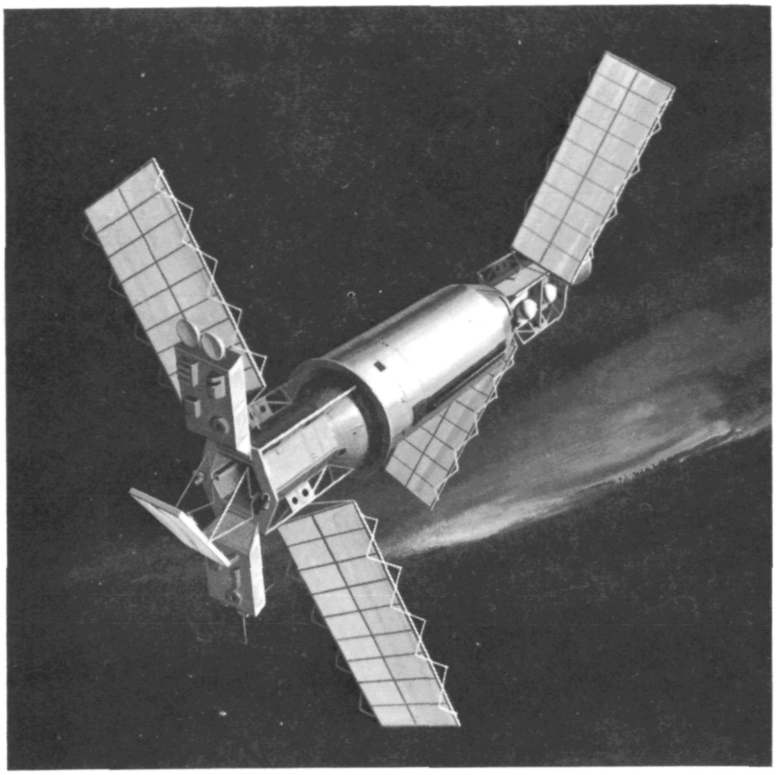


Fig. 1-11 The most famous of the Agenas, the Gemini target vehicle, is shown fixed in space as the astronauts approach for docking and space hitchhike.



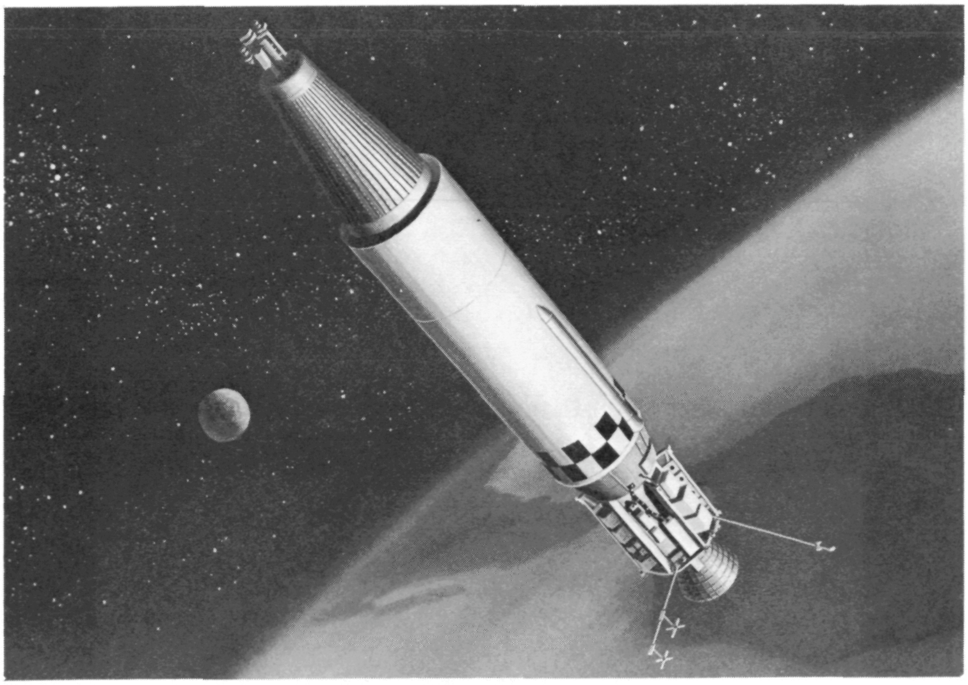


Fig. 1-12 Two Agena spacecraft designs. The upper illustration shows the unfolding payload support rack and solar arrays. The lower picture shows the Snapshot spacecraft, which tested a nuclear reactor in space.

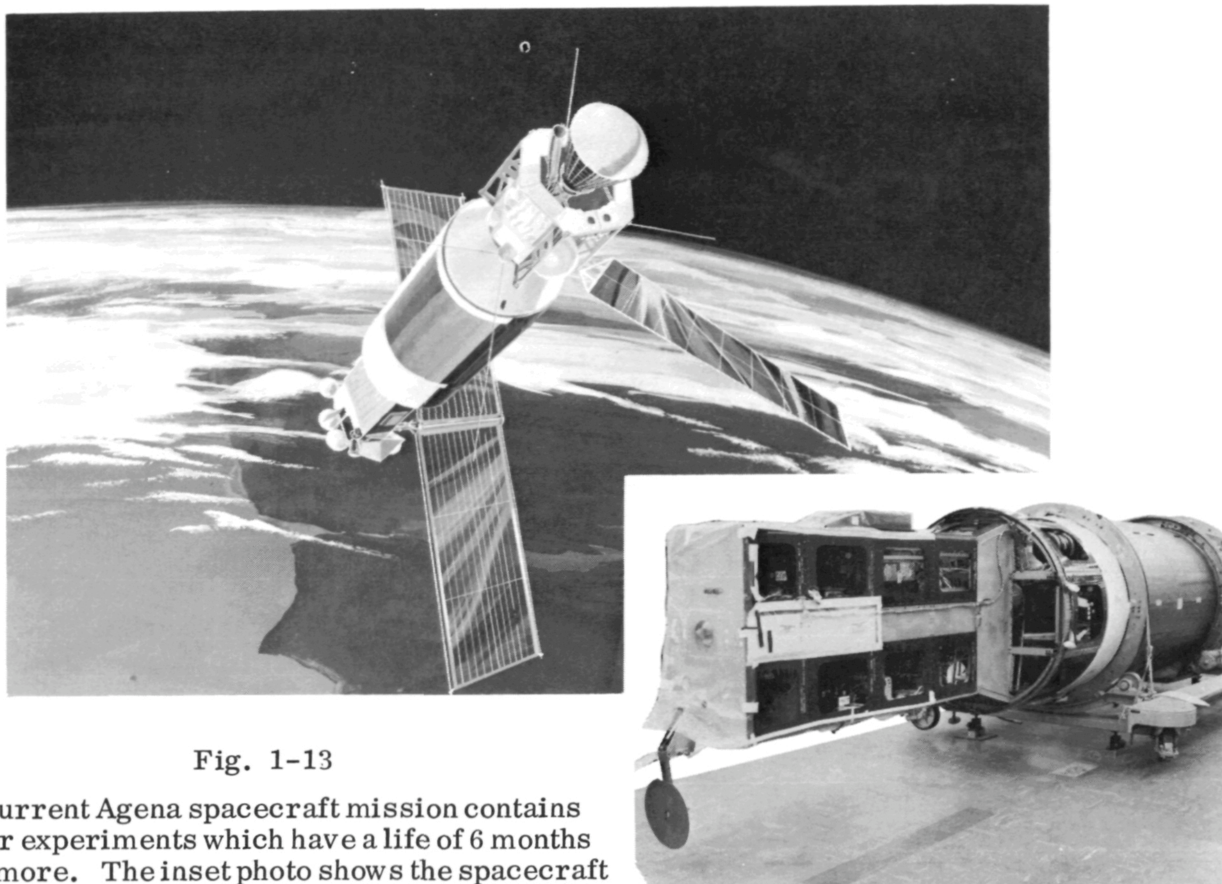


Fig. 1-13

A current Agena spacecraft mission contains four experiments which have a life of 6 months or more. The inset photo shows the spacecraft as it was being readied for testing.

The flexibility to use any of the major boosters and several mission profiles has made the Ascent Agena an important element in unmanned space missions. Operational benefits have been derived because of the Agena's ability to stand at launch-ready condition for up to 15 days on pad awaiting the most opportune launch window. Also, when extended hold has been required on orbit, the Agena has waited for periods up to 20 days prior to restart. The Agena has been a versatile ascent stage for many prominent programs, including:

- Ranger
- Mariner
- Lunar Orbiter
- OGO
- PAGEOS
- ATS
- Nimbus
- SERT
- OAO
- S-27
- DME
- Vela

All were key programs and made important contributions to today's space technology.

Section 2
VEHICLE DESCRIPTION

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The Ascent Agena is composed of five major subsystems: spaceframe, propulsion, electrical power, guidance and flight control, and communications. (In this document, the guidance and flight control and the communications systems are grouped for discussion under the heading "Avionics System.")

The propellant tanks serve as storage vessels for the IRFNA and UDMH propellants and also as an integral load-carrying part of the basic spaceframe.

The vehicle propulsion system consists of a rocket engine, propellant pressurization system, and propellant management system.

The basic electrical system is composed of two batteries, one as a main supply, the other for pyrotechnic power, and a power distribution and control system.

Guidance and flight control is performed by the ascent guidance system. This system is composed of the inertial guidance system (inertial sensor assembly and guidance computer), flight control electronics unit, engine nozzle gimbal, hydraulic actuators, and a cold-gas thruster system (N_2 tank, N_2 regulator, and thrust valves). The system performs vehicle guidance, control, and flight programming functions.

The communications system provides the means by which vehicle data are monitored during system test and transmitted to ground stations during flight operations. This system includes a PCM telemeter, a tape recorder, a baseband assembly, and an S-band transmitter/antenna system.

Major Agena equipment is listed in Table 2-1. Agena weight summary is given in Table 2-2.

Table 2-1

AGENA AND BOOSTER ADAPTER EQUIPMENT

	LMSC Part Number	LMSC Specification
Forward Section		
Battery, Type IVB (2 required)	1462090	1067084
Inertial Sensor Assembly	1460076	1420821
Guidance Computer	1460977	1420821
Flight Control Electronics	1389670-501	1420797
Program Pyro and Monitor J-Box	1389820-501	1420800
PCM Telemeter, Type 4	1460965-1	1420766
UHF Phase Mod Transmitter, Type 19	1460958-1	1420763
Baseband Assembly	1460959-1	1420764
Telemetry J-Box Assembly	1389097-501	1420799
RF Switch, Type 14	1462071-59	1419552
Tape Recorder, Type 29	5507100	1420824
UHF Telemetry Antenna	8100131-503	8100130
Power Distribution and Control J-Box	1389613-501	1420782
Pyro Helium Control Valve	1393028-501	-
Aft Section		
Propellant Isolation Valve	1463144-15	1415273
Nitrogen Regulator	1461952-5	1414273
Thrust Valve Cluster (2 required)	1462553-503	1420741
Hydraulic Power Package	1461704-1	1412972
Pitch Hydraulic Actuator	1461902-11	1067289
Yaw Hydraulic Actuator	1461902-9	1067289
Aft Control and Instrumentation J-Box	1389660-501	1420801
Dual-Start Engine	1462391-1	–
Booster Adapter		
Discrete Control Box	1389034-501	1420781
Self-Destruct Battery	1463150	1415285
UHF Telemetry Antenna	8100131-503	8100130

Table 2-2

AGENA WEIGHT SUMMARY

Subsystem		Weight (lb)
Spaceframe		
Forward Section		86.6
		3.6
		2.8
		15.8
		4.6
		1.2
		19.7
Tank Section		275.9
Aft Section		94.6
		8.3
		1.2
Contingency		5.1
	Total	519.4
Electrical Power		
Primary Batteries		32.2
Power Distribution J-Box		9.1
Aft Control and Instrumentation J-Box		6.2
Main Power Transfer Switch		2.6
Wiring Harnesses		37.4
Pyrotechnic Control Box		4.5
Contingency		0.9
	Total	92.9
Propulsion		
Rocket Engine		225.5
Nozzle Extension Kit		46.0
Dual-Start Kit		7.8
		1.9
Starter Grain Kit (2)		2.6
Starter Igniter (2)		1.2
Prop. Fuel Feed Bellows		1.0
Prop. Oxidizer Feed Bellows		1.0
Helium Fill Coupling		0.3
Pyro Helium Control Valve		3.8
M-69 Pressure Squib		0.1
M-11 Pressure Squib		

Table 2-2 (Cont)

Subsystem		Weight (lb)
Fast Shutdown Kit		1.9
Propellant Vent Coupling		0.4
High-Pressure Helium Coupling		–
Helium Tank		15.9
Check Valve		0.2
Fuel and Oxidizer Feed Bellows (2)		2.7
Propellant Isolation Valves (2)		11.0
Helium Plumbing		2.0
Propellant Plumbing		8.6
Engine Exhaust Shields		3.5
Contingency		3.4
	Total	340.8
Communications		
PCM Telemeter, Type IV		4.0
Base Band Unit, Type 3		2.0
UHF Transmitter		4.0
Telemetry J-Box		5.2
Air-Conditioning Ducting		3.9
Antenna		0.6
RF Switch, Type 14		0.6
Contingency		0.2
	Total	20.5
Guidance & Control		
Flight Control Electronics		8.6
Inertial Sensor Assembly		36.6
Guidance Computer		46.8
Hydraulic Power Package		8.7
Hydraulic Actuators (2)		6.7
Nitrogen Tank		21.2
Nitrogen Regulator		7.9
N_2 Fill Valve		0.3
Thrust Valve Cluster (2)		9.2
Attitude Control Plumbing		2.8
Nitrogen Temperature Probe		0.5
Nitrogen Tank Fitting		0.8
Hydraulic Plumbing		.0
Contingency		1.5
	Total	154.6
Total Vehicle Weight (Empty)		1128.2

2.1 SPACEFRAME

The vehicle consists of three major structural assemblies: the forward, tank, and aft sections, as illustrated in Fig. 2-1. The forward and aft sections provide structure for Agena equipment mounting. The forward section has provisions for mounting the payload (spacecraft and adapter). Most of the Agena systems electrical/electronic equipment is accommodated in the forward section. The propellant tank section consists of an integrally constructed, dual-chambered propellant tank for storage of the fuel and oxidizer used in the rocket engine. This tank is the load-bearing structure in this section of the Agena. The aft section provides support for various equipment of the engine and propulsion, pneumatic attitude control, and hydraulic control systems; for electrical components; and for other related equipment.

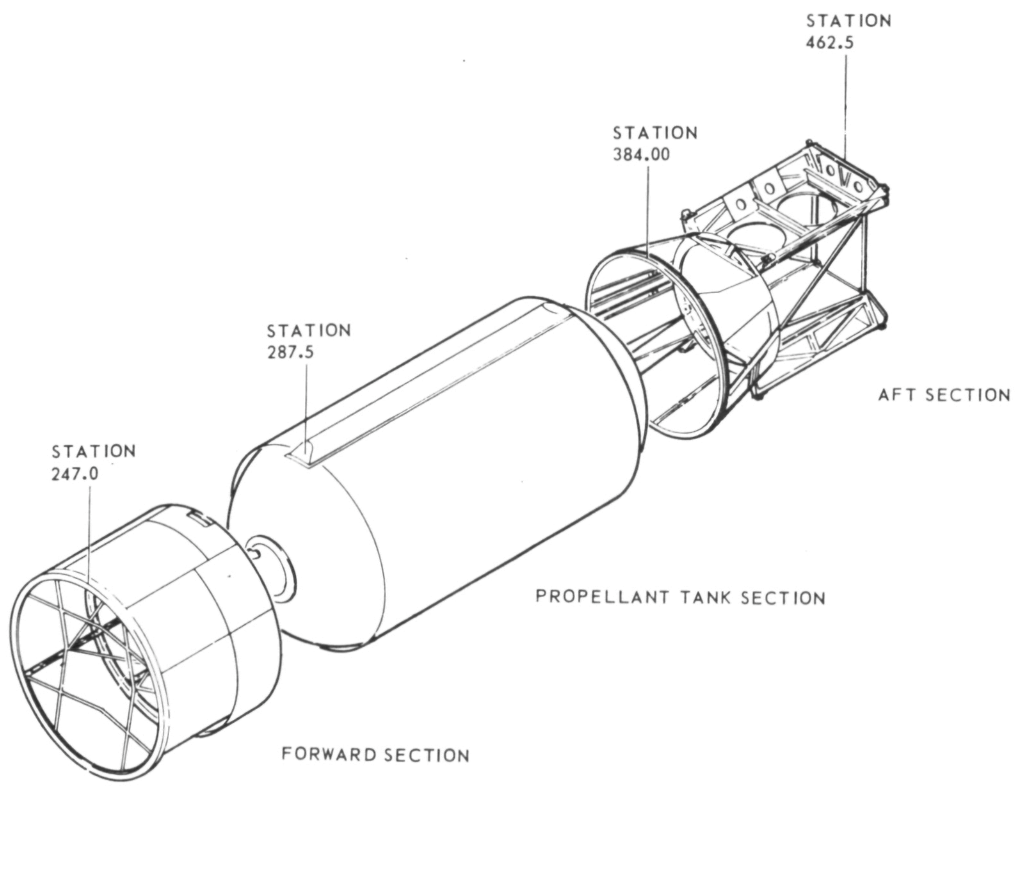

Fig. 2-1 Agena Spaceframe (Basic Structure)

2.1.1 Forward Section

The 40.5-in.-long forward section extending from Station 247.00 to Station 287.5 contains provisions for installation of Agena equipment and mounting and thermal control provisions for payload equipment. It is a semimonocoque structure composed of internal rings and longitudinal members covered by a fixed skin and removable access doors. A welded tubular aluminum truss assembly provides additional strength and facilitates the mounting of equipment. The ends of the tubular truss are pinned forward and aft to eight longerons. The forward hemispherically shaped end of the propellant tank extends approximately 16 in. into the forward section. The aft face of the tubular frame is trussed to fit over and around the tank. The external skin on the forward section, including fixed skin and removable panels, is fabricated from beryllium.

The forward ring of the forward section has eight equally spaced hardpoints with 1/2 in. holes for payload adaptation. A 1/2-in. honeycomb panel with chem-milled aluminum face sheets and hardpoints matching those on the forward ring is available for specific payloads that require additional mounting stiffness. Adaptation of this panel relocates the payload interface plane to Station 246.5.

The forward section mates with the propellant tank section and is joined by countersunk screws and nutplates installed around the circumference of the tank section lapjoint. The nutplates are secured to the inside of the forward section structure at Station 287.00, the forward section/propellant tank interface plane.

With the exception of the propulsion system components, the forward section accommodates the major portion of Agena equipment, as shown in Fig. 2-2. The vehicle electrical umbilical, fuel and oxidizer vents, and helium fill and air-conditioning connections are also located in this section. A controlled temperature/humidity environment is required for ground operation of the inertial sensor assembly, the guidance computer, and the UHF transmitter.

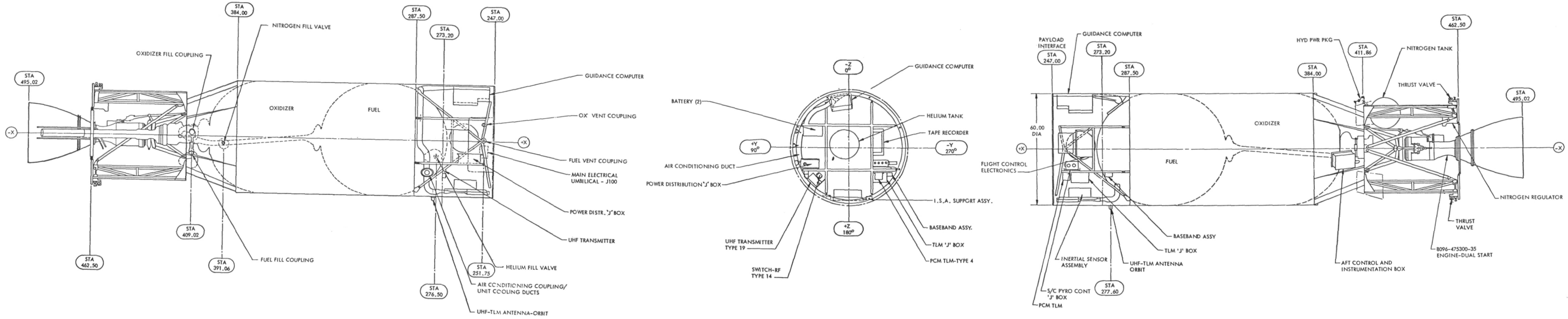

Fig. 2-2 Ascent Agena Inboard Profile

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2.1.2 Propellant Tank Section

The tank section (Fig. 2-3) consists of a dual-chamber propellant tank assembly; the fuel and oxidizer that supply the Agena rocket engine are contained in separate tanks. The assembly is an integral part of the vehicle spaceframe and provides the supporting structure and exterior surface for the center portion of the vehicle. Overall tank length, including the fore and aft hemispherical tank ends, is 129 in. The mechanical interface with the forward section is at Station 287.5; with the aft section, at Station 384.00.

The propellant tank assembly is constructed of welded aluminum sections (Fig. 2-4) forming a cylindrical shell with two compartments. The forward compartment (fuel) is composed of a cylindrical section, two Y-rings, and two hemispherical ends. The aft compartment (oxidizer) is composed of a long cylindrical section, Y-ring, and a single hemispherical end. The two compartments are separated by the aft hemispherical (inner tank bulkhead) end of the forward compartment.

A fuel sump is welded to the aft hemispherical section of the fuel compartment, and an oxidizer sump (Fig. 2-4) is bolted to the aft end of the oxidizer tank. A conically shaped containment screen on top of each sump is fabricated of 0.0035-in.-diameter stainless steel wire, 84 wires per running inch, forming a pore size of 213 microns. The fuel containment sump has a capacity of 0.25 cu ft; the oxidizer containment sump, 0.65 cu ft.

The fuel compartment is pressurized through a boss in the access plate on top of the tank. The oxidizer compartment is pressurized through a fitting on the -Z axis at Station 313.00. Minimum volume of the fuel compartment is approximately 568 gallons. Minimum net volume of the oxidizer compartment is approximately 736 gallons. Nominal operating pressure for the tank is 55 psig, proof pressure is 61 psig, and burst pressure is 69 psig.

2.1.3 Aft Section

The Agena aft section consists of the engine thrust cone and the aft equipment rack, joined into a single assembly. This section supports the rocket engine and provides mounting for other components. The aft section is designed for a booster adapter

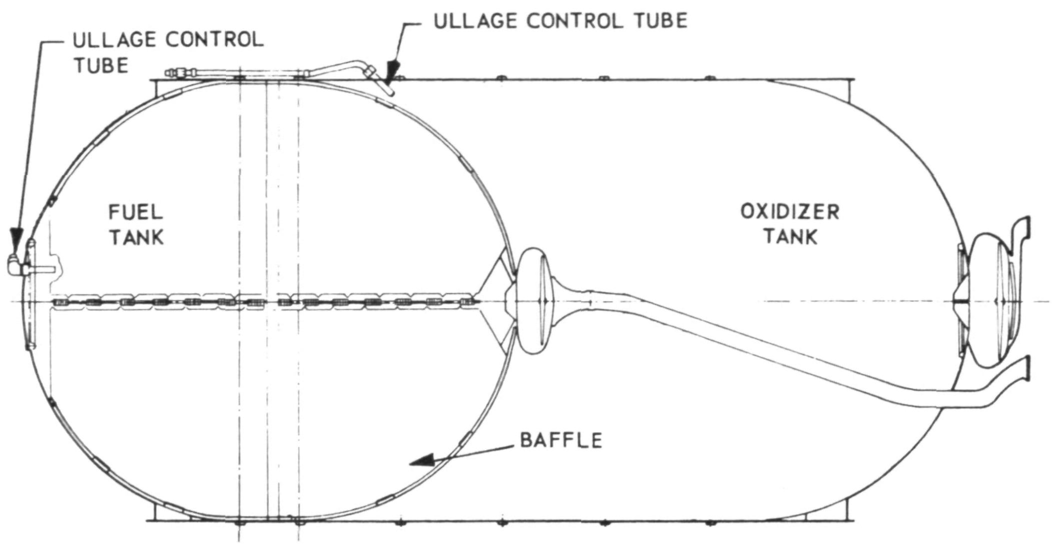


Fig. 2-3 Propellant Tank Section

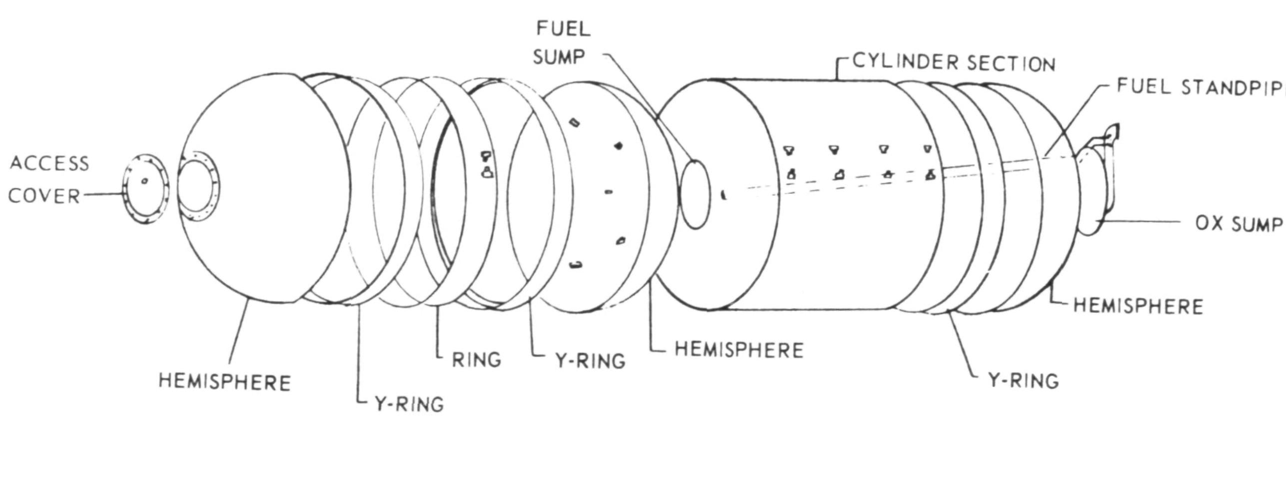


Fig. 2-4 Propellant Tank Details

extending from Stations 383.84 to 462.50. At Station 383.84, the aft section joins the tank section. Station 462.50 constitutes the aft bulkhead.

The engine thrust cone contains a mating ring, which connects the aft section to the propellant tank and booster adapter sections at Station 383.84, and an engine mounting ring, to which the engine and the aft equipment rack are connected at Station 411.86. Longitudinal members join the mating ring to the engine mounting ring and form a truncated cone; openings in the cone structure permit access to the interior.

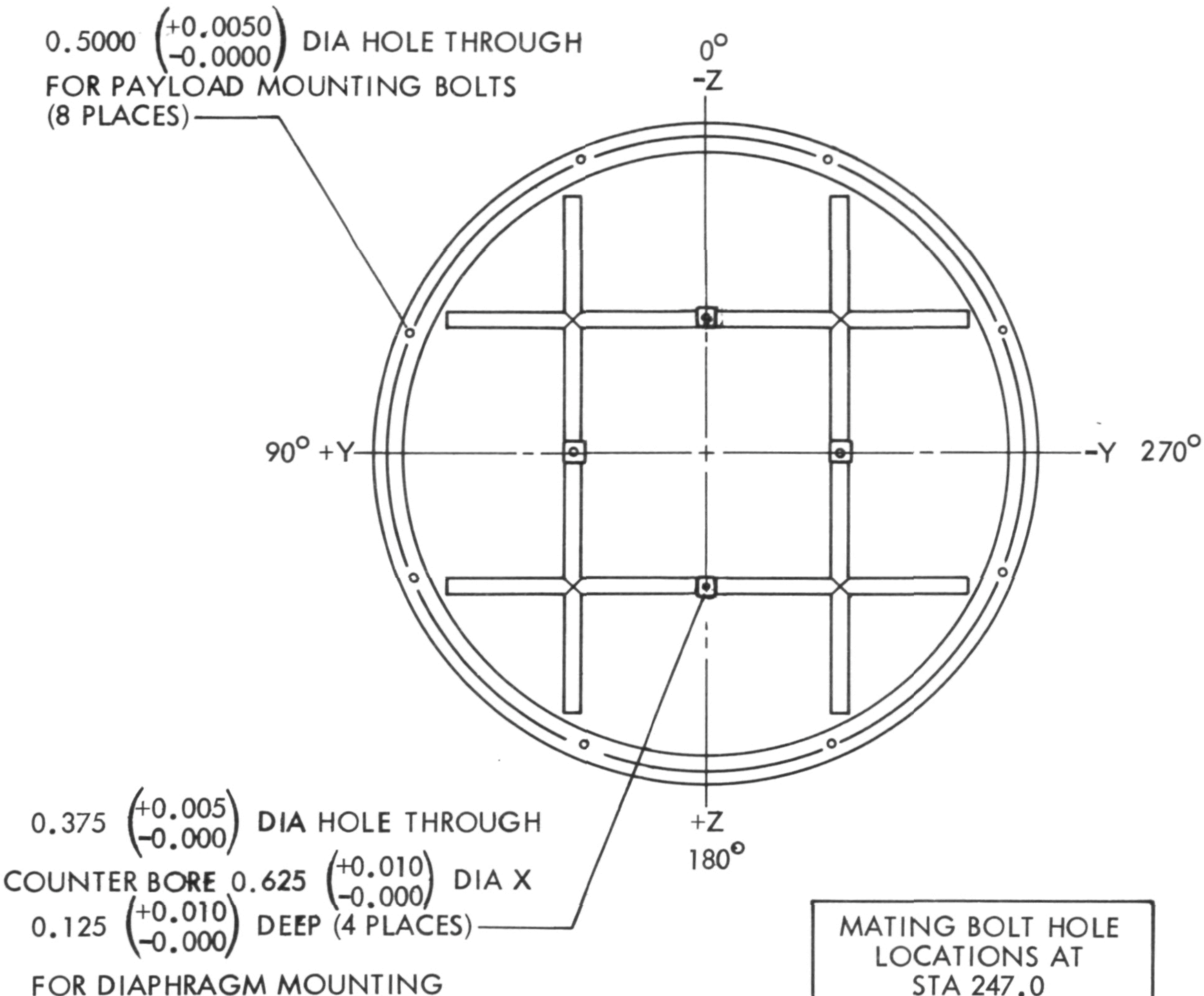
The equipment rack portion of the aft section consists of four truss frames that are cantilevered from the engine cone and attached to the aft bulkhead at Station 462.50. The aft bulkhead consists of two segments located from 340 to 40 deg and 140 to 220 deg. The truss frames and bulkhead sections are structurally augmented by a tubular framework that provides the required lateral and longitudinal structural stability.

In addition to the rocket engine installation, propellant fill and dump quick-disconnects, a nitrogen fill valve, and a hydraulic power package are located on the engine cone. The aft equipment rack supports a gas storage sphere, pneumatic regulator, pneumatic thrust valves, and related hardware. Bracketry installed from Stations 413.00 to 452.00 and between the two support trusses on the -Z axis provide for mounting a gas storage sphere used for pneumatic attitude control. Two pneumatic attitude control thrust valve clusters are mounted at Station 462.50 at 0 and 180 deg.

2.1.4 Major Dimensions and Payload Interfaces

The current Ascent Agena is 60 in. in diameter and 248 in. long with the 45:1 engine nozzle. The length will increase to 266.6 in. when the 75:1 engine nozzle is utilized. Mounting hole payload interface locations at the forward end of Agena Station 247.0 are shown in Fig. 2-5.

Additional detailed Agena dimensioning is shown in Fig. 2-2.



NOTE: 1. RECOMMENDED BOLT FOR FULL STRENGTH:
DIAMETER 0.4995 $\begin{pmatrix}+0.0000\\-0.0010\end{pmatrix}$ AFTER PLATING
2. HEAT TREAT 180,000 - 200,000 PSI ULTIMATE TENSILE STRENGTH
3. TORQUE 480-690 IN.-LB

MATING BOLT HOLE LOCATIONS AT STA 247.0	
DEGREES	RADIUS (IN.)
22°29'59"	29.006
67°30' 9"	29.009
112°29'55"	29.010
157°29'49"	29.010
202°29'33"	29.010
247°29'24"	29.009
292°29'33"	29.006
337°30' 3"	29.007

Fig. 2-5 Payload Mounting Interface

2.2 PROPULSION SYSTEM

The propulsion system provides the vehicle impulse that results in the velocity increment (or increments) necessary to perform intermediate-stage booster or orbital vehicle missions. The Ascent Agena has sufficient propellants for a total burn time of 240 sec, which can be divided into one, two, or three burns of various durations to achieve the orbit or orbits dictated by the mission requirements.

The propulsion system consists principally of the rocket engine subsystem with its control equipment and the propellant tank pressurization and the propellant management subsystems necessary to support the rocket engine operation (Fig. 2-6). The propulsion system description in this volume is presented within this basic breakout of propellant pressurization, propellant management, and rocket engine subsystem elements.

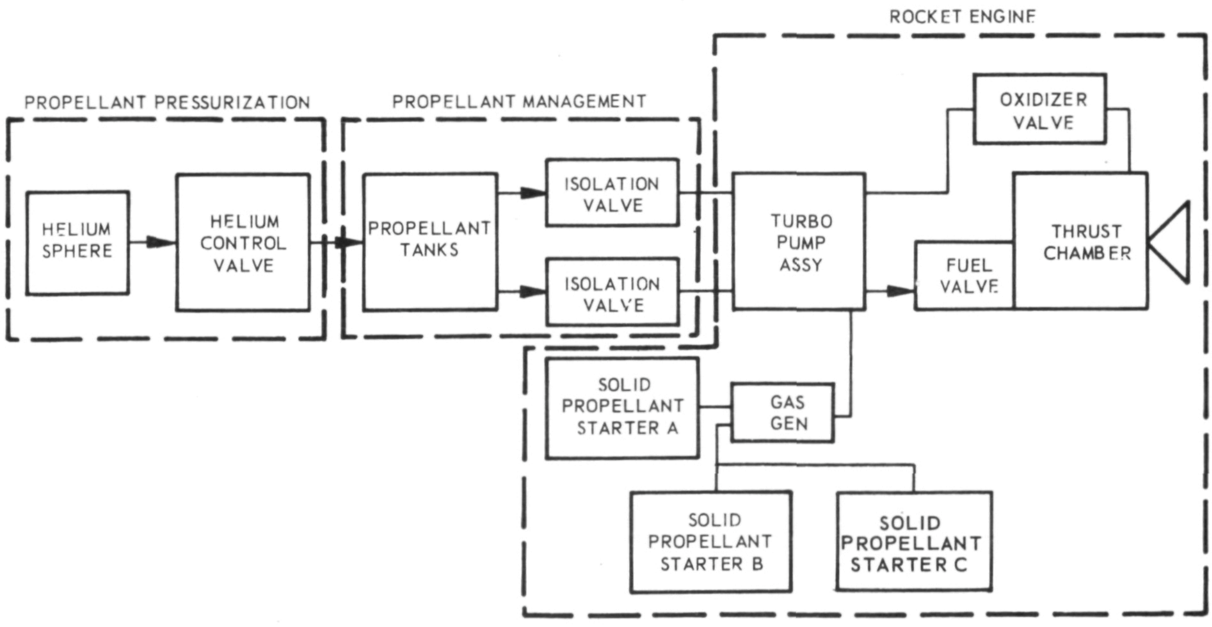


Fig. 2-6 Propulsion System Block Diagram

Propellant Pressurization. The propellant pressurization system receives, contains, regulates, and distributes high-pressure helium gas to pressurize the propellant tanks to ensure correct propellant flow by suppressing pump cavitation. The system includes provisions for conducting launch base checkout and servicing operations.

Propellant Management. The propellant management system receives propellants from the ground support equipment (GSE) loading system. Propellants are routed to the respective tank cells, where they are stored until consumed by the engine. An important function of the management system is performed by containment and scavenging sumps. These devices assure a propellant supply to the pump inlets for the initial and subsequent engine starts and enable more complete depletion of the usable propellants in the tanks. An optional propellant dump system may be installed to vent residual propellants left in the tanks after engine operation is complete.

Rocket Engine. The rocket engine is the USAF Model YLR81-BA-39 (Bell Aerosystems Co. Model 8096) liquid bipropellant power package, which uses inhibited red fuming nitric acid (IRFNA) and unsymmetrical dimethylhydrazine (UDMH). The engine is regeneratively cooled by oxidizer that passes through drilled passages in the thrust chamber walls prior to entering the injector. The engine is mounted in a gimbal ring, a device resembling a universal joint that allows the engine to move laterally and vertically to control the vehicle during engine operation. Hydraulic actuators supply the motive force for thrust chamber movement in response to signals produced by the vehicle guidance system. The rated thrust of the engine, in a vacuum, is 16,000 lb, and the total burn duration is 240 sec. Test stand firings for as long as 600 sec have been made with no engine deterioration. Single-, dual-, or triple-start capability is provided by installation of the appropriate starter assembly.

2.2.1 Propellant Pressurization

The basic function of the pressurization system is to maintain sufficient helium in the propellant tanks to prevent propellant pump cavitation and ensure that engine pressure and flow requirements are met. Minimum tank pressures are dictated by the minimum allowable inlet pressures at the fuel and oxidizer pumps. Tank pressures must be sufficiently high so that the combination of feedline pressure drops and acceleration-produced

pressures will result in inlet pressures above the level at which major cavitation can occur. Propellant pump cavitation can lead to pump discharge pressure decay and premature engine shutdown.

The structural design of the propellant tanks imposes upper limits on the allowable tank pressures. The tank pressures cannot be too high, but they must be high enough to provide the structural stiffness essential for support of the vehicle forward rack and payload weight during liftoff and high-g load periods.

2.2.1.1 Operational Characteristics. The basic orifice-controlled pressurization system (Fig. 2-7) is used for both single- and multistart missions. The heart of the system is the pyro helium control valve (PHCV), which is an assembly of three fixed-orifice isolation valves: one for each propellant tank and one for the helium storage tank. Each isolation valve has two positions: the helium supply valve has one position for filling the sphere from a ground supply and one for in-flight pressurization. The propellant valves have one position for tank vent and one for pressurization.

The PHCV is actuated and the pressurant bottle is completely blown down for the first burn, regardless of the number of firings or the burn time sequence. Consequently, the pressure history in the propellant tanks will vary from mission to mission. This fixed-orifice concept provides overall system simplicity and reliability, but it requires that all operational parameters be carefully surveyed and optimized to work safely and effectively.

The operational features of the PHCV are illustrated in Fig. 2-8. In the closed (pre-flight) position, the propellant tanks and the high-pressure helium supply are isolated from each other. The vent lines from the propellant tanks are routed through the PHCV to the vent couplings, and the helium fill line is routed through the valve to the pressurant sphere. Squib pressure cartridges are fired to actuate the PHCV and open the flow passages from the sphere to the propellant tanks. This actuation, through related poppet and piston mechanisms, simultaneously isolates the propellant vent and helium fill couplings from the propellant and pressurant tanks. Actuation of the PHCV occurs with firing of Squibs 1 and 2, 1.5 sec after Agena first-burn ignition. Subsequently, additional pressure cartridges are fired to actuate the oxidizer pressure

isolation valve (OPIV) to the closed position. This isolates the oxidizer tank from the helium source and the rest of the pressurization subsystem. Otherwise, mixing of fuel and oxidizer vapors might occur when the blowdown has been completed and pressures throughout the system have equalized.

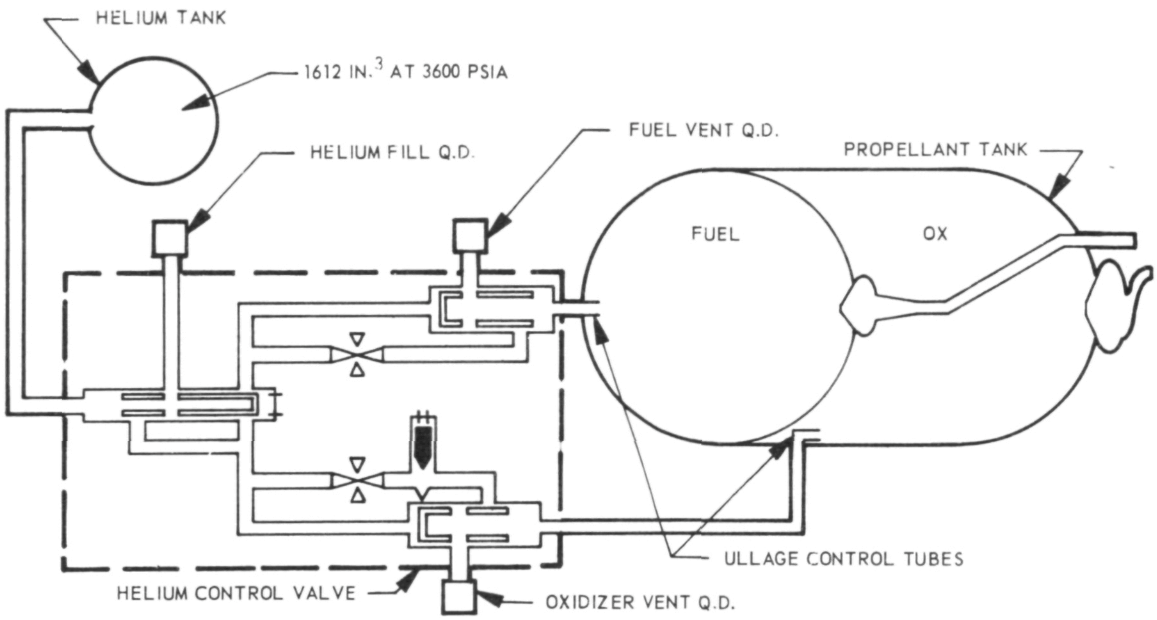


Fig. 2-7 Propellant Pressurization System

As previously indicated, the gas pressures in the oxidizer and fuel tanks must be sufficient to produce engine pump inlet pressures that will suppress pump cavitation. The minimum steady-state engine pump inlet pressures are approximately 12 psia for fuel and 8 psia for oxidizer. While the engine can run satisfactorily at these inlet pressures, propellant system boilout and vapor lock conditions may exist during pump refill and engine restart operation. Thus, minimum tank pressures of 22 psia and 14 psia for fuel and oxidizer, respectively, must be available at the start of second burn to assure vapor-free pump and feedline operation. A maximum tank pressure of 55 psig is dictated by the tank design working pressure and must not be exceeded.

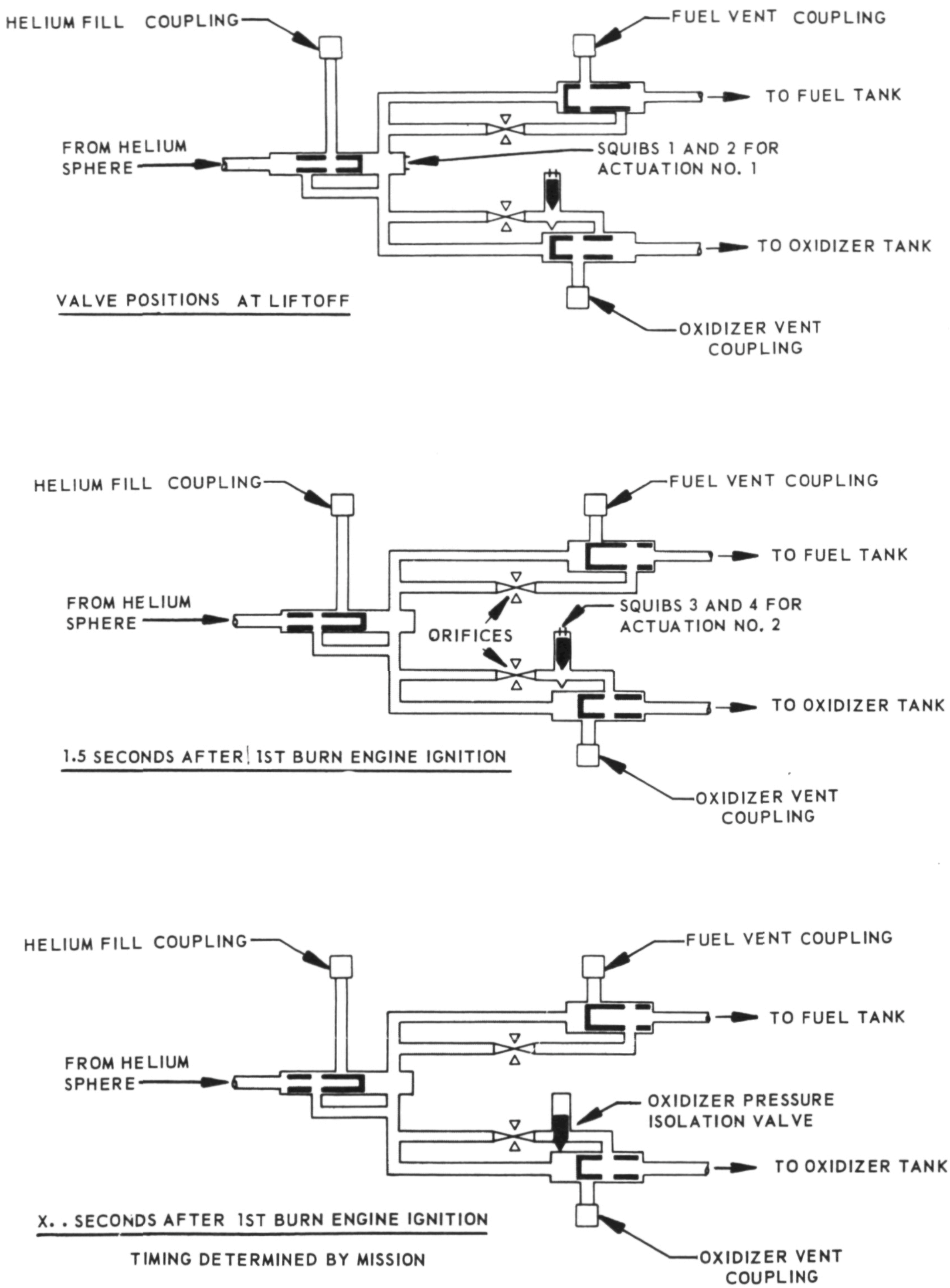


Fig. 2-8 Pyro Helium Control Valve Operation

The propellant tank imposes a further limitation in that pressure on the oxidizer side of the common bulkhead must not exceed that on the fuel side. Otherwise, bulkhead reversal may occur, giving rise to a possible bulkhead failure and consequent mixing and explosion of the hypergolic propellant.

Propellant acceleration pressures cannot be used in planning pressurization system operation because of the system characteristics and the acceleration range of 0 to 7 g for a typical mission. Therefore, ground-loaded ullage pressures in the two propellant tanks and control of tank gas pressures during propellant withdrawal from the tanks during engine operations must provide for positive differential at all times.

The pressure in the propellant tanks drops sharply for the first 1.5 sec of engine startup because propellant is being withdrawn without pressurization. When the sequence timer signals opening of the PHCV and pressurant flows to the tanks, the pressure increases for a few seconds and then once again starts to decay. This small pressure spike that occurs upon actuation is slightly more pronounced for the fuel tank because of the smaller volume, and thus provides an added measure of protection for the bulkhead.

The timing of the PHCV opening is very important; early or late opening could cause loss of the vehicle through ruptured propellant tanks or premature engine shutdown. Too early an opening before sufficient propellant had been withdrawn would cause a rapid pressure rise and probably rupture the tanks through overpressurization. If it is opened too late, the tank pressure will be so low that pump cavitation and engine shutdown will occur. Opening the PHCV 1.5 sec after the engine signal represents a compromise in timing that produces the most efficient operation and desirable propellant tank pressures.

The oxidizer pressure isolation valve (OPIV) isolates the oxidizer tank from the rest of the pressurization system to prevent the equalizing of pressure in the helium sphere and the fuel and oxidizer tanks with possible vapor exchange. This valve is an integral part of the HCV, actuated by pressure cartridges that are independent of the PHCV first-actuation cartridges. For single-burn vehicles, the OPIV generally performs

its isolation function at the time of the engine shutdown backup signal. The small quantity of remaining helium is allowed to flow to the fuel tank before propellant dumping takes place. The oxidizer tank would show a commensurately lower pressure at blowdown. On multistart vehicles, it is desirable to utilize all available helium to assure adequate pressures at both pump inlets prior to second-burn ignition. This is necessary to provide complete suppression of all vapor bubbles and elimination of possible vapor lock in the pumps at refill prior to second burn. It takes approximately 390 sec for all useful helium pressure to transfer to the tanks, at which time the fuel and sphere pressures begin to approach each other. At this point, the OPIV is sequenced closed by signal from the sequence timer. Generally, the multistart vehicles have higher tank pressures at the end of 240 sec of elapsed burn time than do the single-burn vehicles because of the above and of coast-period heat transfer. Additional heat transfer from the tanks to the colder helium tends to heat the propellant during the coast period of a dual burn, causing expansion and subsequent pressure increase.

The single-actuation characteristic of the PHCV imposes a limit on the minimum first-burn period for which satisfactory performance can be provided within the allowable pressure-time histories and structural constraints of the tanks. Dual-burn periods of 150/90, 220/20, 235/5 sec and triple-burn periods of 123/92/25 sec are typical. A study of tank pressure-time histories (Figs. 2-9 through 2-12 indicates that as the first-burn period is shortened, with all other parameters constant, the tank ullage volumes stop increasing at shutdown and the propellant tanks increase in pressure. A minimum allowable first-burn period exists where propellant tank pressures will not exceed the allowable pressure maximum of 55 psig after all available helium has been transferred prior to second burn. Any mission requirements for a first-burn period of approximately 100 sec or less must be evaluated to determine if the basic pressurization system configuration and operating parameters can provide satisfactory conditions.

2.2.1.2 Component Descriptions. The helium fill coupling (Fig. 2-13) provides access to the vehicle pressurization sphere for loading and unloading high-pressure helium gas. This coupling must seal against helium leakage from liftoff until PHCV operation shortly after Agena first ignition. The coupling also provides a means of filling the

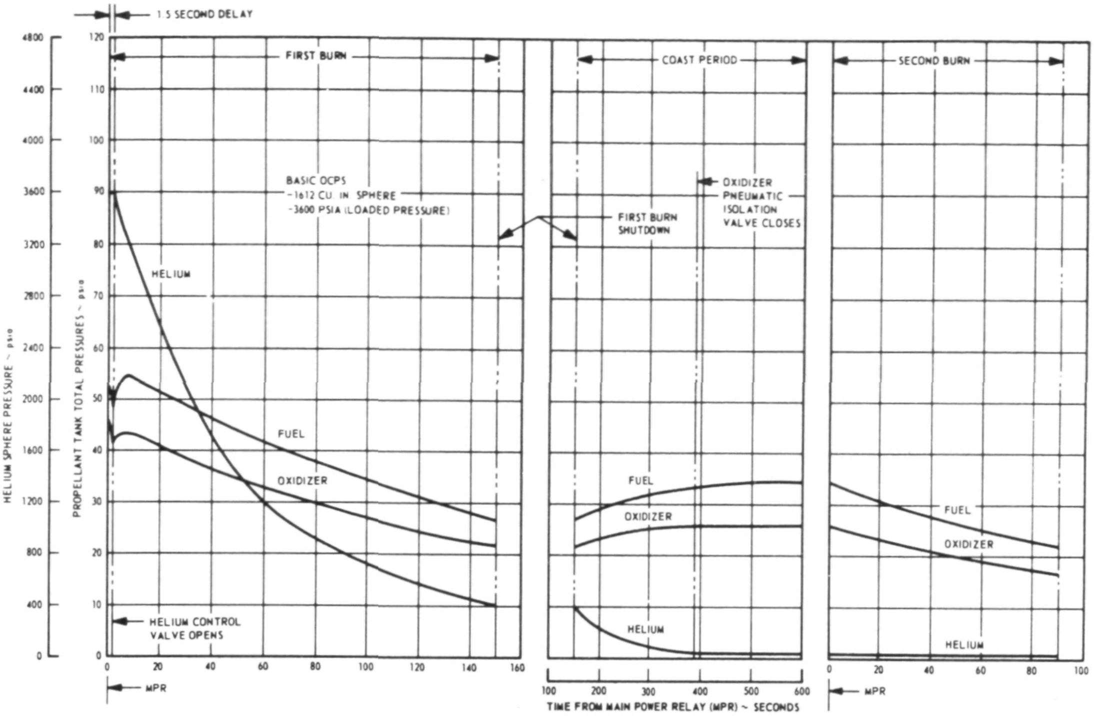


Fig. 2-9 Pressurization Operating Parameters (150/90 Burn Period)

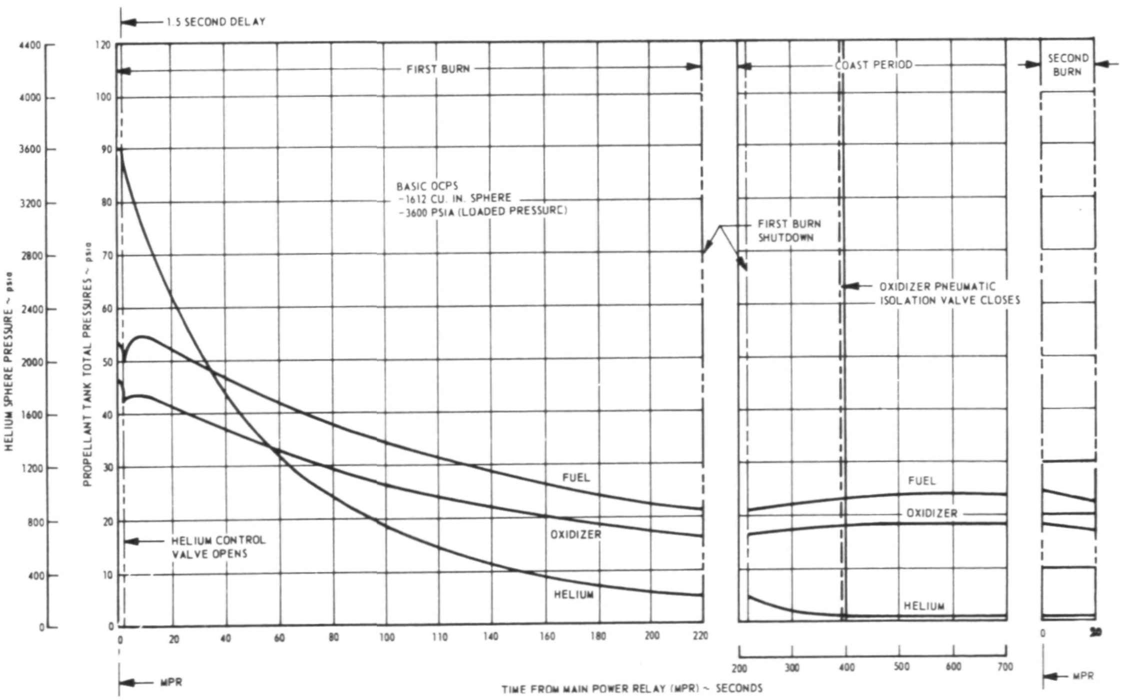


Fig. 2-10 Pressurization Operating Parameters (220/20 Burn Period)

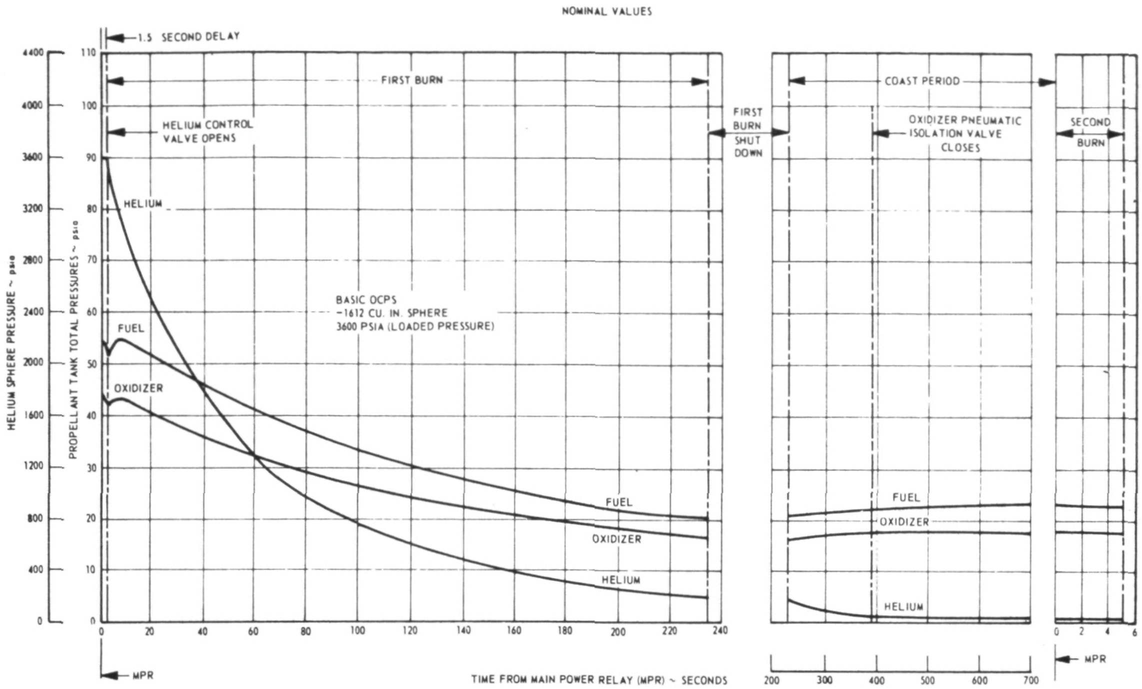


Fig. 2-11 Pressurization Operating Parameters (235/5 Burn Period)

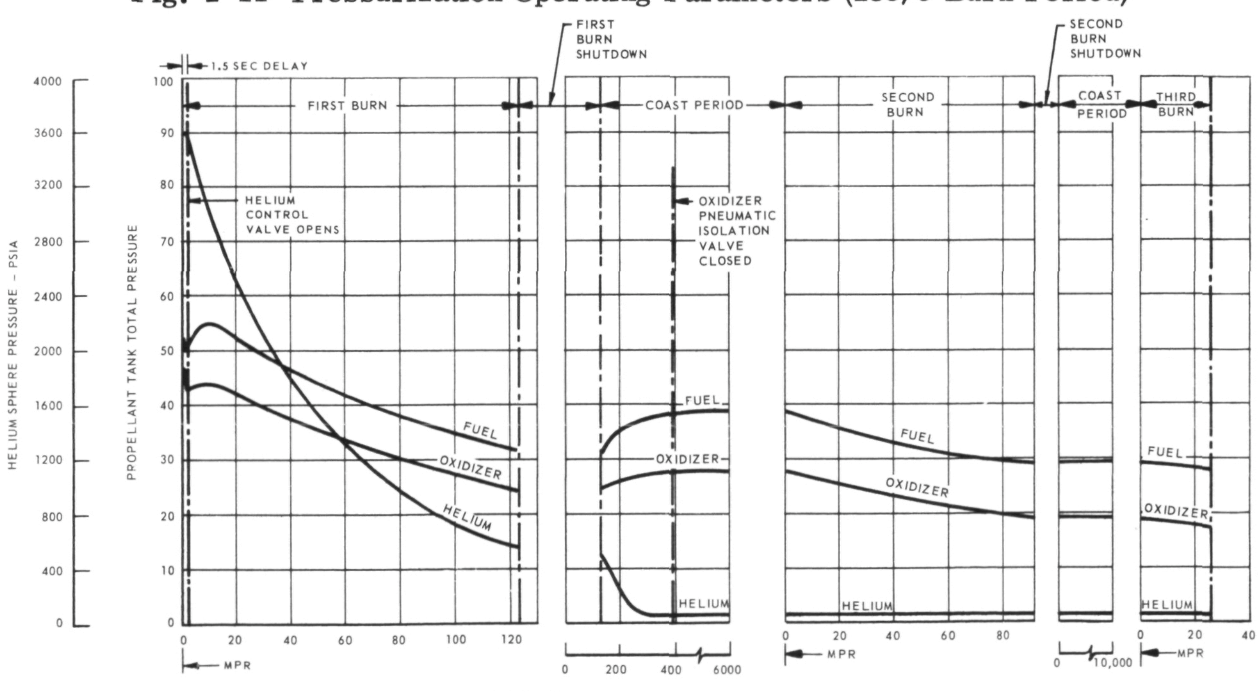


Fig. 2-12 Pressurization Operating Parameters (123/92/25 Burn Periods)

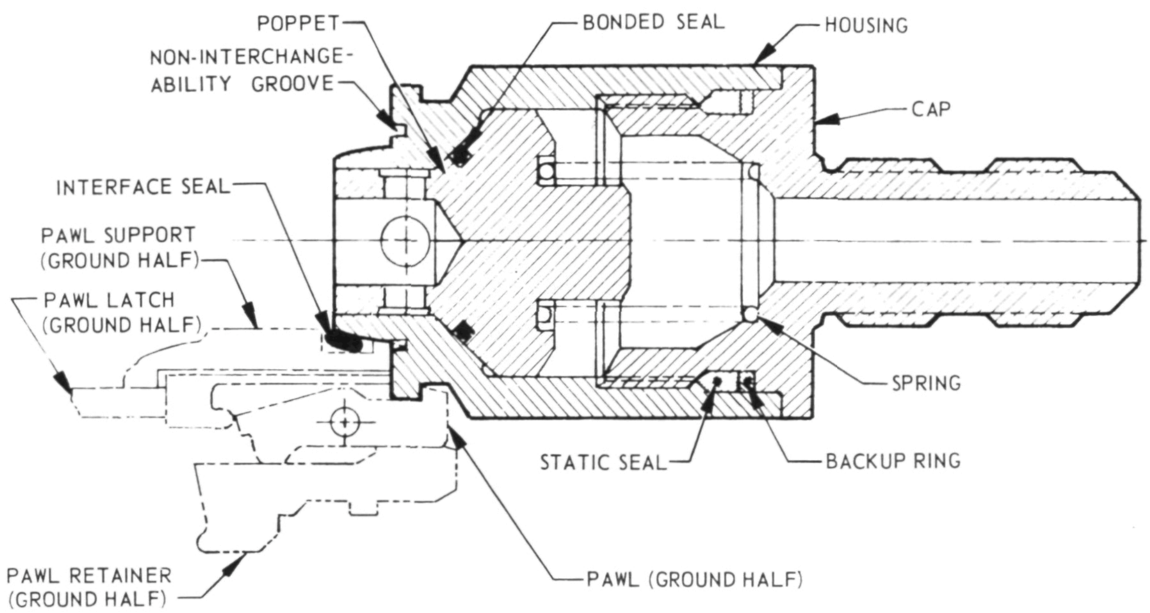


Fig. 2-13 Helium Fill Coupling

high-pressure system during checkout and keeping a positive pressure in the system to prevent contamination. The flight half of the helium fill coupling is a normally closed, spring-loaded, poppet-type check valve which is actuated to the open position by a probe in the mating ground half of the coupling. When the flight and ground halves are initially mated, the flow passages in both units are in the closed position. These passages are opened with a separate 300-psig nitrogen gas source that is applied to the ground half coupling pneumatic actuator. This actuator opens an inner flow valve and extends a probe into the flight half. The probe pushes the flight-half poppet off its seat and permits flow through the coupled units. A locking mechanism, consisting of spring-loaded pawls on the ground half, grips a flange on the flight half to hold the two units together. Uncoupling takes place at vehicle launch when tension is applied to the ground-half uncoupling lanyard by a pneumatic actuator. The design of the coupling is such that should the lanyard fail to uncouple the locking mechanism, the tips of the locking pawls shear away without impairing the flight-half sealing surfaces. A mechanical noninterchangeability ring on the mating surface of both halves prevents mating the attitude control system ground-half fill coupling with the helium fill flight half.

The helium storage tank is loaded with high-pressure helium gas from the umbilical system prior to launch. The helium gas is stored at or below $120^{\circ}F$ until it is used in the propellant tanks to replace the propellants burned by the engine. In the basic vehicle configuration, a 1612-cu-in. tank pressurized to 3600 psia is used for both single- and multistart missions. A special kit for vehicles of certain programs provides a 900-cu-in., 3600-psia tank for use with single-burn missions. The smaller tank allows installation of extra equipment around the tank mounting area. Some optional configurations even place the small tank in the aft section for better forward section space utilization.

All gas supply tanks on Agena vehicles are made of two titanium alloy hemispheres welded together. Each hemisphere has a threaded port machined into it; one of these ports is utilized for filling and dumping. A thermocouple temperature measurement device is attached as basic equipment to monitor the helium gas temperature. The maximum allowable tank temperature $165^{\circ}F$ during loading and 120° after top-off. Helium tank pressure may be optionally monitored at the helium control valve.

The pyro helium control valve (PHCV) is constructed primarily of aluminum alloys. A 10-micron, stainless-steel filter is located upstream of the flow control orifices to filter the helium supplied from the sphere. The static and dynamic seals on the fuel side are made of UDMH-resistant butyl rubber; those on the oxidizer side are of IRFNA-resistant Viton "A" compound. The stainless-steel, 0.023-in.-diameter replaceable orifice inserts are compatible with the 3600-psia tank pressure used in most vehicle configurations. Any special program requirement specifying the use of initial pressures lower than 3600 psia requires a commensurate change in orifice size. If the initial tank pressure is decreased, the orifice size must then be increased to maintain the gas flow at the proper rate during the critical period following engine first-burn ignition. The PHCV has ports to accommodate transducers for monitoring helium and propellant tank pressures. The transducers are optional equipment; when they are not installed, the ports are plugged.

The primary function of the propellant tank vent couplings is to provide a means of venting and pressurizing the propellant tanks during propellant loading and preflight tank ullage pressurization operations. The couplings also provide access to the tanks for system checkout of tank pressurization and for applying the tank storage and holding pressures that minimize system and propellant contamination. The couplings serve no purpose after launch other than to seal against leakage of gas, liquid, or vapors. The flight halves of the propellant vent couplings (Fig. 2-14) are normally closed, spring-loaded, poppet-type check valves that are actuated to the open position during the mating operation by a probe in the mating ground half. When mated, the two halves are mechanically locked together; the flow passages through both couplings are then in the full-open position. The locking mechanism consists of spring-loaded pawls that grip a flange of the flight half. Uncoupling takes place at vehicle launch, when tension is applied to the ground-half coupling disconnect lanyard by a pneumatic actuator. Propellant-compatible elastomer sealing materials are used on the poppet seat and as a static interface seal to prevent leakage between the body and the fitting end of the flight half. The design of both the vehicle and ground halves is such that should the lanyard system fail to release, the locking feature of the ground half will cause the pawls on the ground half to shear through the flight-half flange without impairing the effectiveness of the sealing surfaces. A mechanical noninterchangeability ring that prevents mating of the wrong ground-half coupling is included in the mating surfaces of both halves.

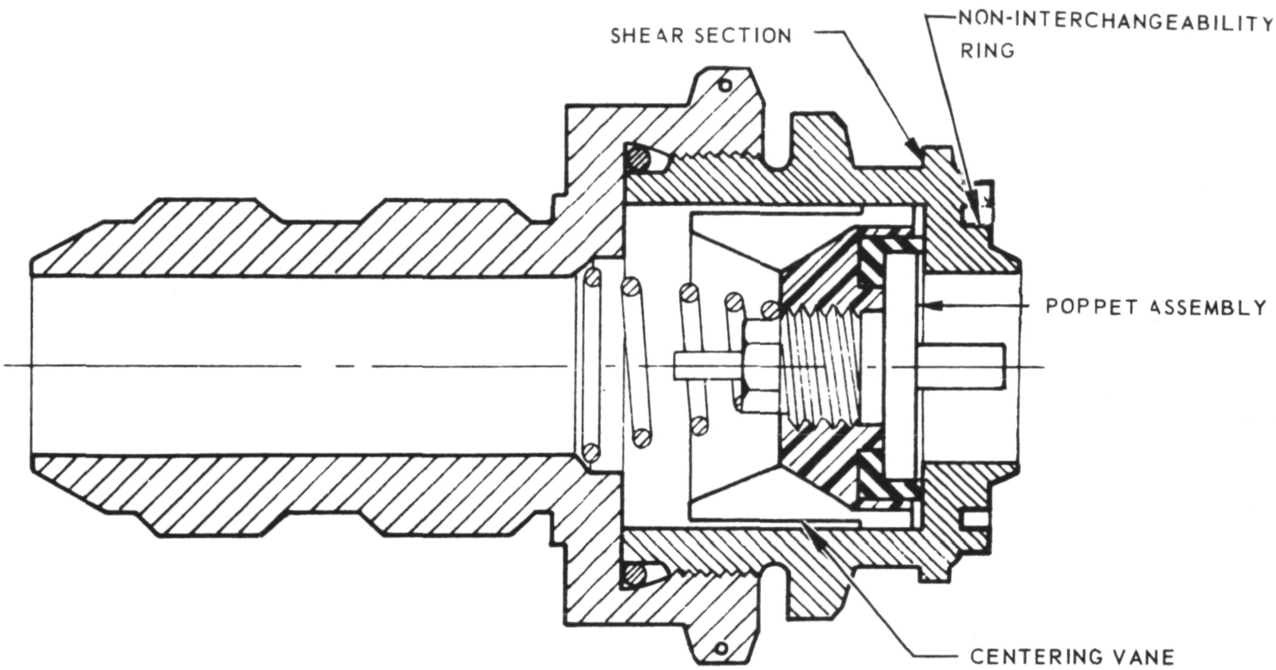


Fig. 2-14 Airborne Half of Propellant Tank Vent Coupling

The ullage control tubes are small pipes inserted into the top of each propellant tank cell. By sealing off the vent system after a certain propellant level has been reached, the tubes limit the amount of propellant which may be loaded and thereby provide the proper ullage. When the tanks are loaded with propellants, about one percent of the tank volume is left void to accommodate propellant expansion and to provide a gas reservoir to suppress pressure fluctuations. Minimums for each tank are 0.9 cu ft for oxidizer and 0.75 for fuel. The fuel ullage control tube is mounted in the fuel tank manhole cover; the oxidizer tank tube is welded into the intermediate Y-ring that separates the fuel and oxidizer cells.

2.2.2 Propellant Management

The term "propellant management" is used here to describe the functions and related actions of systems and units that control Agena propellants from launch through the propellant dump phases of flight. Propellant management deals with the propellant tank and propellant feed systems (Fig. 2-15), including the tank containment-scavenging sumps and the propellant isolation valves. The sumps hold a quantity of gas-free propellants to assure containment of the proper quantity of propellants for reliable engine start. The sumps permit operation without a large quantity of unneeded reserve propellants, minimize unusable residual propellants, and assure passive propellant containment between starts.

2.2.2.1 Propellant Containment Sumps. The need for propellant containment and scavenging sumps (Fig. 2-16) is dictated by propellant instability during low-gravity coast periods that occur between engine firings. On the launch pad and during the boost phase of flight, the ullage gas is located in the forward end of the tank. During the first coast following boost, however, the Agena is essentially in free fall and experiences only the acceleration produced by aerodynamic drag. This drag produces an aft-directed acceleration on the vehicle which attempts to buoy the ullage bubble aft toward the sump inlets and start the propellants migrating forward, away from the engine feed lines. Also, during low-altitude orbit coast periods that follow first burn, drag forces or attitude control forces might be sufficient to cause adverse propellant migration. The sump must be designed to hold liquid against these adverse forces in sufficient quantity to prevent significant gas ingestion throughout the engine restart transient period.

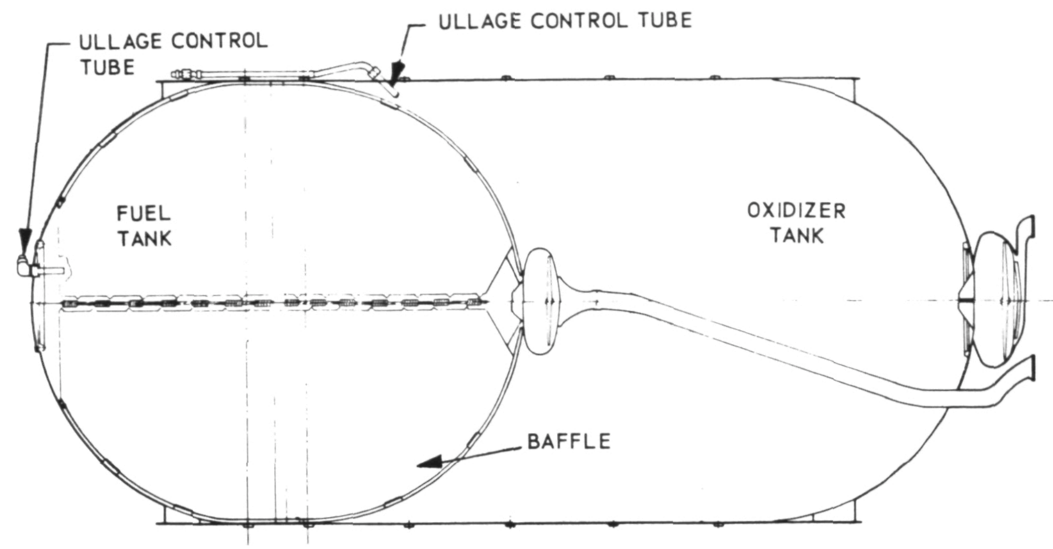


Fig. 2-15 Propellant Tank and Sumps

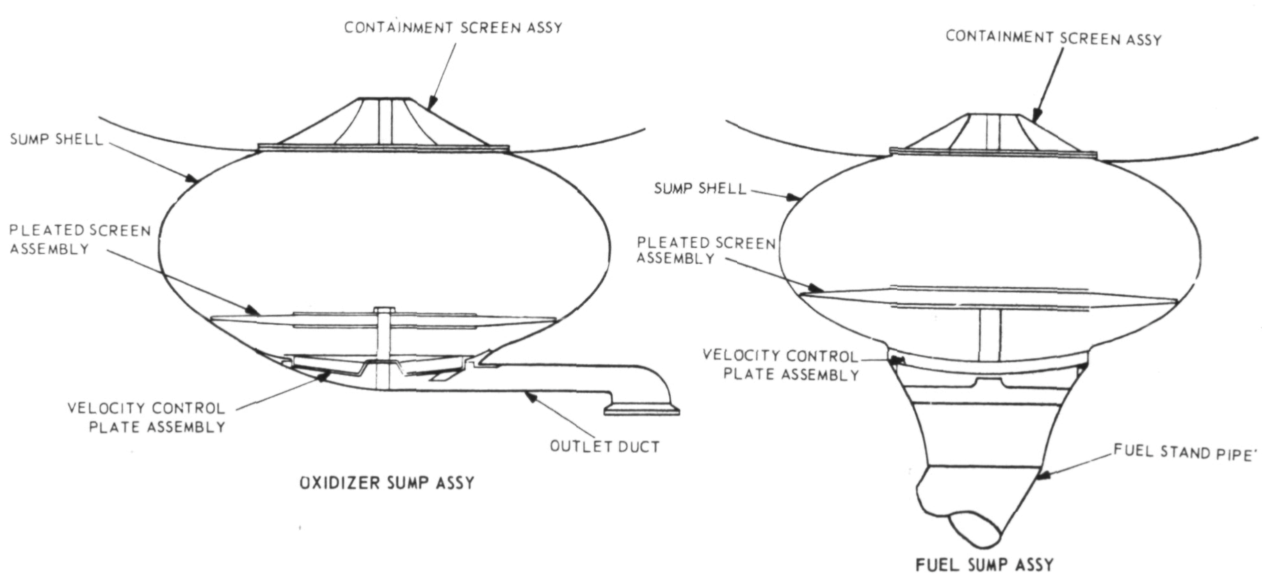


Fig. 2-16 Containment/Scavenging Sump Assemblies

In general, there are four phases in the engine restart transient flow-time history. The first phase consists of opening propellant isolation valves located at the sump outlet, and filling the feed lines and engine which were vented down after the previous firing. Initially, propellant movement is induced by tank pressure and subsequently augmented by turbine pump spinup. In the second phase, the thrust chamber oxidizer valve opens and flow begins through the chamber cooling passages and out of the injector. Up to this point, fuel is not flowing out of the engine and the turbine pump is being driven only by the solid-propellant starter. The third phase begins as the gas generator valve opens and liquid propellants begin to flow into the gas generator combustion chamber. These liquids mix and ignite, increasing turbine speed until the solid-starter grain is exhausted. The final phase starts when the fuel valve opens. This is followed by turbine speed maximum overshoot, starter grain burnout, and eventual speed decay toward steady-state conditions.

Sump Design. The fuel and oxidizer sumps shown in Fig. 2-16 satisfy these requirements. They are constructed of formed aluminum sections welded together, differing primarily in size and method of attachment to the tank. The fuel sump is welded to the fuel tank; the oxidizer is attached to the oxidizer tank with screws. Each sump consists of a structural assembly, a containment screen, a pleated screen, and a velocity control plate. The fuel pump permits a steady flow rate of approximately 136 gpm and a peak flow rate of 160 gpm. The oxidizer sump permits an average flow rate of 182 gpm and a peak flow rate of 215 gpm. The containment screen at the inlet end of the sump provides for surface-tension containment of propellants within the cavities of the sumps at all times. The pleated screen prevents or delays the flow of pressurization gas into the engine. Propellant flow velocity distributions are controlled by the velocity control plate mounted at the outlet end of each sump. The sumps contain approximately 0.25 and 0.67 cu ft of fuel and oxidizer, respectively.

The unusual geometry used on the oxidizer sump outlet is due to the aft rack and engine cone space envelope. Sump operational parameters require that the sump be located in the center of the tank, while the engine pump inlet flanges are offset and cannot be relocated without engine redesign. With the propellant isolation valves, pump inlets, and sump all located in the same small area, the right-angle design of the oxidizer sump outlet duct provides the only means of aligning and connecting the oxidizer sump to its related pump inlet.

Materials and characteristics of the barrier screens used in both sumps are identical and, like the sumps, differ only in size and shape. The screens are made of stainless-steel wire that is processed by heating under pressure. The heating and pressing process is specified to accurately size the screen holes and to assure uniformity under all stresses applied to the screens by propellant motion. Using stainless-steel wire for the screens offers excellent strength and corrosion resistance.

Containment Screen. One of the principal elements of the sumps is the containment screen. This screen, a truncated cone located at the inlet to each sump, uses surface tension to contain propellant against acceleration disturbances. These disturbances may appear laterally if pitch, yaw, or roll attitude corrections are made, or longitudinally if either forward-or aft-directed accelerations are experienced. The surface-tension forces of the small radius screen mesh adequately handles all expected disturbances.

During first engine start, there is a small time increment of turbine exhaust duct thrust without propellant withdrawal from the sumps. Inasmuch as the duct thrust is small, the vehicle weight high, and the time increment small, any ullage bubbles in the vicinity of the containment screen will not be displaced significantly during this period. After initiation of propellant flow from the tank, the hydrodynamic drag on the bubbles near the screen quickly exceeds their buoyant force and they are drawn to the screen. Upon contact, they distort and wrap around the screen, causing increased propellant flow velocity through the unblocked portion of the screen. The velocity increase reduces the pressure downstream of the screen, a process which continues until the pressure differences across the screen exceed the screen's capillary hold capacity and gas begins to be ingested by the sump. Since this whole process occurs at a rapid rate, it is likely that some gas may enter the sump in the few cases when vehicle weights, coast times, and altitudes combine to produce sufficient drag force.

The mesh of the containment screen cone is sized so that the hydrostatic head difference between the apex and base of the cone exceeds the gas-holding capability of the screen under vehicle acceleration introduced by main engine thrust. At about the same time

as the gas exceeds the screen capillary-hold capacity and penetrates the screen, the engine thrust chamber ignites. The thrust-produced buoyant forces then move and trap the gas against the containment screen. During first burn, as propellant is forced aft through the peripheral areas of the sump screen, trapped gas is released from the containment screen apex until the propellant surface under the screen has moved forward to a point where the hydrostatic head difference is supported by the gas-holding capability of the screen at that acceleration level. As engine operation continues and vehicle weight decreases through propellant consumption, the containment screen continues to release gas. At vehicle first shutdown, a small quantity of gas may remain trapped under the containment screen. The amount of gas trapped depends on the terminal acceleration of the vehicle and even in the worst case is not detrimental to restart performance.

As liquid is withdrawn from a vessel, in this case the sump, a depression is produced in whatever free surface exists. Carried to the extreme, the surface can be depressed into the outlet of the vessel and result in the withdrawal of gas along with the liquid. This phenomona is called surface dip. Surface dip becomes more pronounced as the gravity field or liquid level is reduced or as the pumping rate is increased.

Surface dip penetration and pressurization gas ingestion into the sump outlet is delayed significantly by sump design and internal surface-tension barriers. A design having the sump inlet and outlet on the same centerline and having the inlet larger than the outlet produces a dip of reasonably shallow depth which is arrested by the barrier screens and control plates until the propellants in the sumps drop below the level of the velocity control plate. The pleated screen assembly, located a few inches below the containment screen, is the first screen that the surface dip encounters on its way to the sump outlet. This screen tends to damp and redistribute areas of high velocity, and hence high pressure drop, to areas of lower velocity and flow resistance. To penetrate the screen, the pressure difference across the screen must exceed the surface-tension forces. As long as the local inertia forces do not exceed the surface-tension forces, gas penetration will not occur; and the dip will contact the screen and spread out over it.

Velocity Control Design. Just below the pleated screen is the velocity control plate assembly, consisting of a screen and perforated plate. This assembly distributes the flow and thus lowers the naturally high velocity in the sump outlet ducting. The oxidizer sump is more critical in this respect because of its right-angle duct at the sump outlet. Lowering the velocity of the propellants at the velocity control plate assembly also reduces peak velocity at the pleated screen, thereby increasing its bubble-arresting capacity. Oxidizer residuals benefit from lowering of the local inertia forces at the inlet to the right angle duct, which postpones the start of local surface dip into the duct and hence the oxidizer pump. Propellant isolation valves are located in the sump discharge duct to minimize the effect of heat soak-back from a hot engine. Following an Agena firing, this heat transfer vaporizes the propellant in the feed lines and propellant pumps. The growth of these vapor pockets could push the liquid out of the sumps, leaving them filled with gas for the next engine start. To avoid this, the isolation valves are closed upon engine shutdown, which simultaneously vents the feed lines to the vacuum of space. The heat soak-back and related boilout effects thus are reduced to a minimum, and adequate liquids can be retained in the sumps for refill of the engine pumps and lines before the next engine ignition.

Scavenging Process. A major function of the containment/scavenging sumps is to minimize the residual propellants – those remaining in the tanks and lines above the propellant isolation valves at engine cutoff. More complete use of the propellants in the tanks (reducing residuals) directly increases the vehicle payload capability.

In performing the propellant scavenging function, each of the sump parts perform a task that aims toward prevention of gas ingestion but at the same time is effective for propellant scavenging. The sump containment screen as well as the pleated screen and velocity control plate resist propellant slosh and rotation tendencies at the tank exit. This effectively prevents vortexing induced by fluid rotation during withdrawal of propellants. These features are important for effective propellant scavenging in that more fluids can exit at the tank before gas ingestion begins. Gas ingestion starts soon after the propellant surface level passes below the pleated screen. Thus, engine shutdown is initiated by gas ingestion as a result of the passage of the propellant surface below the velocity control plate. Gas ingestion is prevented as long as propellants completely cover the pleated screen.

Oxidizer pump scavenging capability is diminished and resultant oxidizer residuals are somewhat larger than for fuel residuals as a result of the right-angle outlet duct required by the offset position of the oxidizer pump. Fluid withdrawal is more turbulent with an attendant loss of sump scavenging efficiency.

Restart Reorientation Process. The size of the sumps, particularly the oxidizer, is determined by the amount of propellants necessary to start and run the engine long enough to produce reorientation and refill, plus a reasonable amount to account for engine performance deviations, trapped gas from first-burn shutdown, and other contingencies.

The oxidizer withdrawal rate relative to sump volume is much higher than for the fuel. Therefore, it determines the minimum reorientation time. In the oxidizer sump, the propellant level reaches the pleated screen approximately 1.74 sec after engine ignition during a nominal engine start transient with an initially full sump. To assure minimal gas ingestion, the leading edge of the reorienting oxidizer must contact the containment screen prior to the 1.74 sec. The leading edge of the reorienting propellant strikes the containment screen in less than 1.74 sec from engine start; therefore, both the oxidizer and fuel sumps begin to refill before the internal liquid level recedes to the pleated screen, thus satisfying the sump refill leading-edge fall-time requirement. Given a nominal engine start transient and realistic onboard oxidizer weights of from 110 lb to 4000 lb, approximately 2 sec is required for completion of propellant orientation. Thus, for approximately the first 5 sec of engine operation, the propellant reorientation process is occurring. Immediately after this the propellant undergoes sloshing, resulting from the decay of initial rebounds of the reorienting propellant.

As long as the containment screen remains covered during the rebound-slosh process, the refill will occur normally; and the rebound-slosh effect on refill is negligible. There is, however, a possibility that for low amounts of reorienting oxidizer, the resultant rebound and side-to-side slosh motion will increase residuals by preventing a few pounds of oxidizer from entering the sump before shutdown.

2.2.2.2 Propellant Feed and Load Equipment. The propellant feed and load equipment (Fig. 2-17) consists of two propellant fill couplings, two propellant isolation valves (PIVs), two feed bellows that compensate for any misalignment between the engine and tank, and two load bellows that connect the propellant fill couplings to the propellant isolation valves and to the engine and tanks when the valves are open. The fuel and oxidizer feed and load equipment is essentially the same except that there is a fuel-return boss in the load bellows into which the fuel dumps when it returns from the integrated hydraulic package. The feed bellows are attached directly to the pump inlets by bolts and nuts and are sealed by a Teflon gasket. Marman clamps attach the bellows to the propellant isolation valves. The fill bellows are attached to the PIV valves and couplings with Marman clamps.

Fill Couplings. The propellant fill couplings (Fig. 2-18) provide access to the vehicle fill lines through which the propellants are loaded into and unloaded from the tanks. Subsequent to vehicle launch, these couplings must prevent propellant leakage. The propellant fill coupling is a normally closed check valve that is actuated to the open position by a probe in the mating GSE ground-half coupling. When the vehicle half and the GSE half are initially mated, the propellant transfer passages in both units are in the closed position. To open these coupling propellant passages, the ground-half coupling pneumatic actuator is pressurized to 150 psig with nitrogen gas. This opens the ground-half coupling directly and also extends the probe that pushes the poppet in the vehicle half off its seat, thus permitting flow through the coupled units.

The two halves are mechanically locked when mated by a mechanism incorporated in the ground-half coupling. The locking mechanism uses a ball-detent with a spring-loaded sleeve. Uncoupling the flight and ground halves takes place during vehicle launch when the locking mechanism is released by a force applied through a lanyard to the locking sleeve on the ground half. This force is provided by a GSE pneumatic actuator during the initial liftoff sequence.

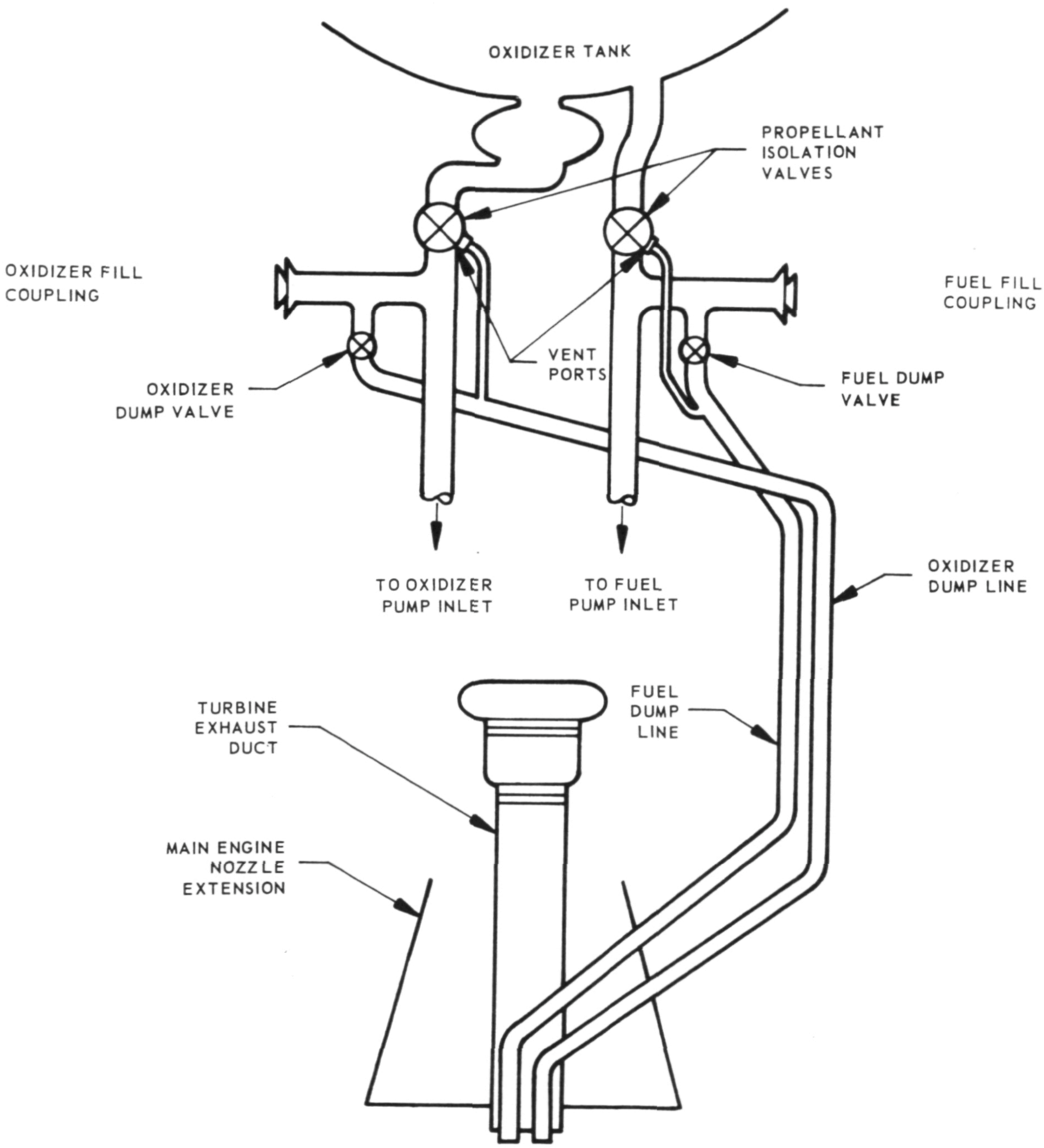


Fig. 2-17 Feed, Load, Vent, and Dump Equipment

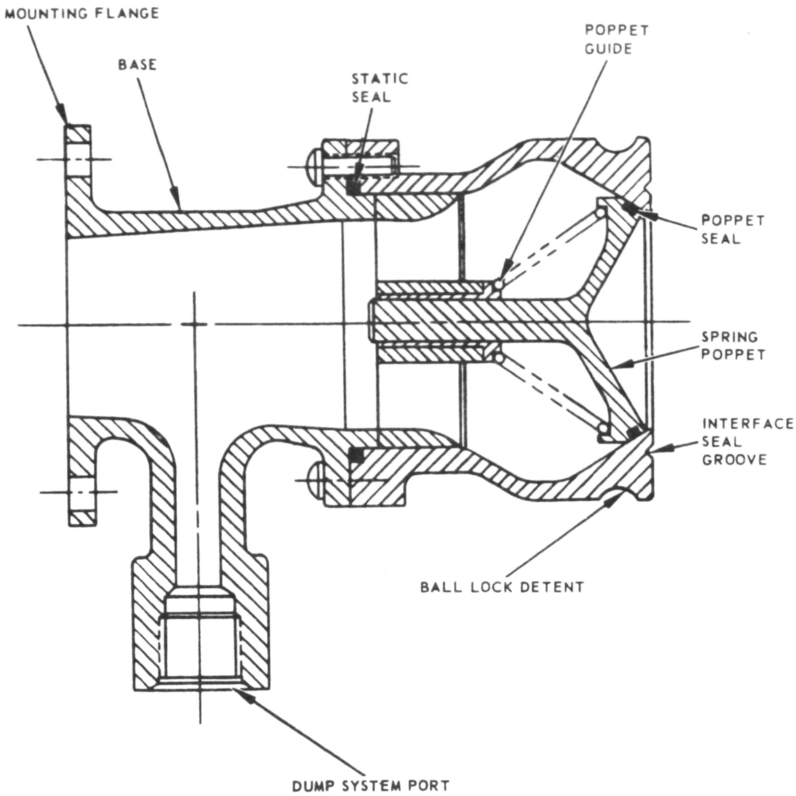


Fig. 2-18 Airborne Half of Propellant Fill Coupling

Propellant Isolation Valves. The PIVs are motor-operated blade valves located between the propellant tank outlets and the propellant pump inlets to prevent propellant boilout from occurring in the sumps. In the open position, these valves allow filling, draining, and flushing of the tanks and engine through an integral port located on the engine side of the blade. When closed, the PIVs isolate the turbopump inlets and fill lines from the propellant sumps and tanks and also vent the engine entrapped residual propellants through integral vent ports which open only after the valves have sealed the tank/engine interface. Post-burn thrust from these vented propellants, in combination with engine tailoff impulse, produces a forward directed vehicle acceleration that aids in preventing propellant deorientation for up to an hour, depending on orbit altitude and vehicle weight. The vented propellants are routed through lines that are parallel to the turbine exhaust duct. The ends of the vent lines are aligned through the vehicle center-of-gravity to reduce the attitude of disturbances due to venting thrust. The PIVs are open at liftoff, closed shortly after engine first-burn shutdown, and reopened 2 sec before second-burn ignition.

The multiburn vehicle incorporates a propellant isolation valve for each sump. Each PIV and a small bellows with suitable attaching hardware serve as the connection between the sump and engine pump inlets. Single-burn vehicles replace the PIVs with a longer bellows assembly that connects the fill line, feed line, and engine pump together.

2.2.2.3 Propellant Dump Equipment. The propellant dump equipment is an option that provides for dumping propellants and venting the vapors in the tanks subsequent to final engine burn. Propellant dumping prevents extraneous torques on the vehicle during orbital life. Such torques may result from leakage of pressurization gas, liquids, or vapors from the tanks. Also, by venting both propellants, the possibility of tank bulkhead reversal due to leakage in the fuel system is prevented. Separate and independent dump equipment is provided for fuel and oxidizer. Propellants are dumped by means of a pyro-operated dump valve (Fig. 2-19) connected to approximately 10 ft of tubing that extends along the turbine exhaust duct. This same tubing is also used for venting propellants from the engine when the propellant isolation valves are closed.

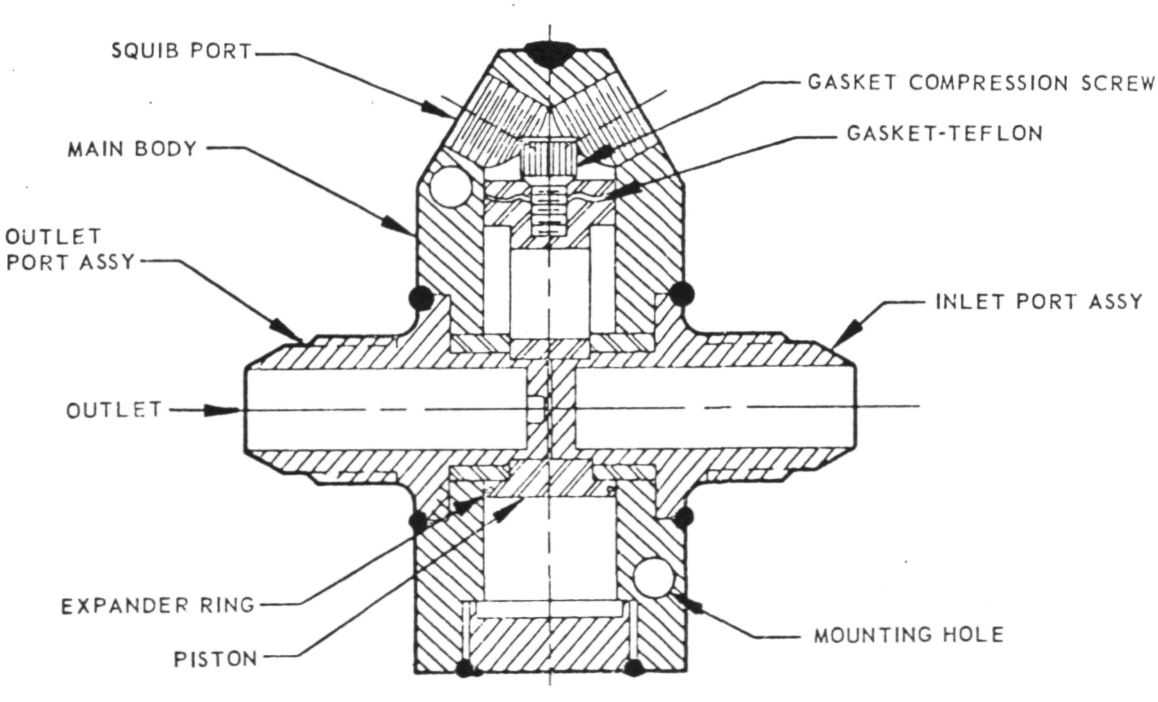


Fig. 2-19 Fuel or Oxidizer Dump Valve

The dumping of residual oxidizer is initiated at final engine shutdown by opening the pyro-operated oxidizer dump valve. Approximately 100 sec later, depending on the previously set fuel dump time delay, the fuel dump valve is fired open by the sequence timer and fuel dumping begins. The fuel dump delay provides assurance that propellant tank bulkhead reversal will not occur during the dumping process. When either dump valve is opened, liquid begins to flow into the dump line and continues until only a few pounds of liquid remain in the tank and feed lines. At this time, liquid flow ceases and the flow into the dump tube becomes a mixture of helium and propellant vapor. Gaseous flow continues for some time until all remaining liquid propellant has been evaporated and tank pressure becomes ambient.

2.2.3 Rocket Engine

The rocket engine (Fig. 2-20) is a Bell Aerosystems Company Model 8096 (USAF designation YLR 81-BA-11) that is turbopump-fed with liquid propellants. The engine's single combustion chamber is regeneratively cooled by oxidizer that passes through drilled passages in the thrust chamber walls and throat prior to entering the injector; the expansion nozzle is cooled by radiation. The engine is mounted in a gimbal ring that allows the engine thrust vector to be varied for pitch and yaw control during engine operation. Hydraulic actuators supply the motive force for thrust chamber movement in response to signals sent by the ascent guidance system.

The propellant pumps are geared to a single turbine that is driven by gases from a gas generator in which the propellants are reacted in a fuel-rich mixture ratio. A duct installed along the engine nozzle exhausts the turbine gases overboard.

The engine performance parameters of thrust, mixture ratio, and specific impulse are closely controlled by cavitating venturies in the pump manifolds and in the gas generator flow circuit.

The main propellant valves are incorporated in the engine. The oxidizer valve is spring loaded and is operated directly by the oxidizer pressure buildup or decay. The fuel valve is also spring loaded; however, its operation by fuel pressure is controlled by a solenoid valve that actuates when oxidizer manifold pressure builds up.

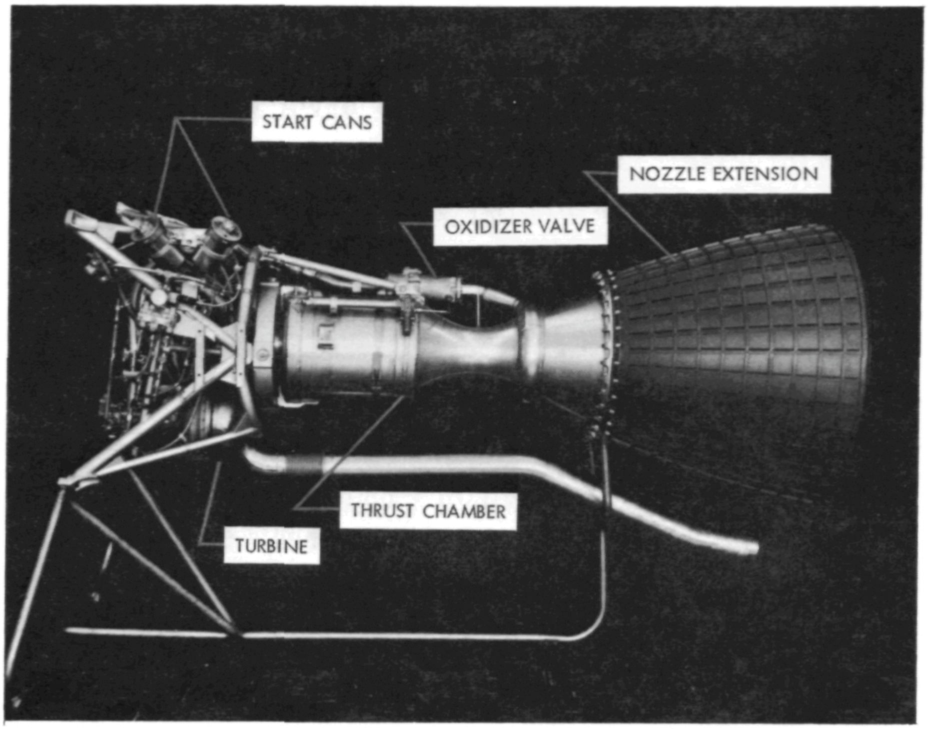


Fig. 2-20 Agena Rocket Engine

Engine start is initiated by a single command signal energizing the gas generator solenoid valve and igniting a solid-propellant start cartridge that provides sufficient hot-gas flow to the turbine until pump action makes the system self-sustaining. From one to three start cartridges can be installed, providing for single- to triple-start capability. Engine shutdown is initiated by closure of the gas generator and fuel solenoid valves when their electrical power is terminated. As the gas generator turbine slows, oxidizer pressure decays and the oxidizer valve then closes. The start and shutdown sequences ensure that oxidizer flow precedes and overlaps fuel flow for reliable operation. A single-operation, fast-shutdown oxidizer valve kit is available as an option for reducing the oxidizer postflow loss on multiple-burn missions. In this kit, a small volume of compressed gas is used to quickly close the oxidizer valve prior to oxidizer pressure reduction.

For convenience, the engine components are described in six basic groups that lend themselves to a sequential presentation compatible with normal engine start and operation. The component descriptions and the subsequent discussion of engine operational sequence should provide an adequate understanding of engine operation. The engine component groups are as follows:

- Solid propellant starter
- Turbopump assembly
- Gas generator assembly
- Thrust chamber assembly, main propellant valves, and pressure switches
- Engine mount and gimbal assembly
- Electrical control equipment

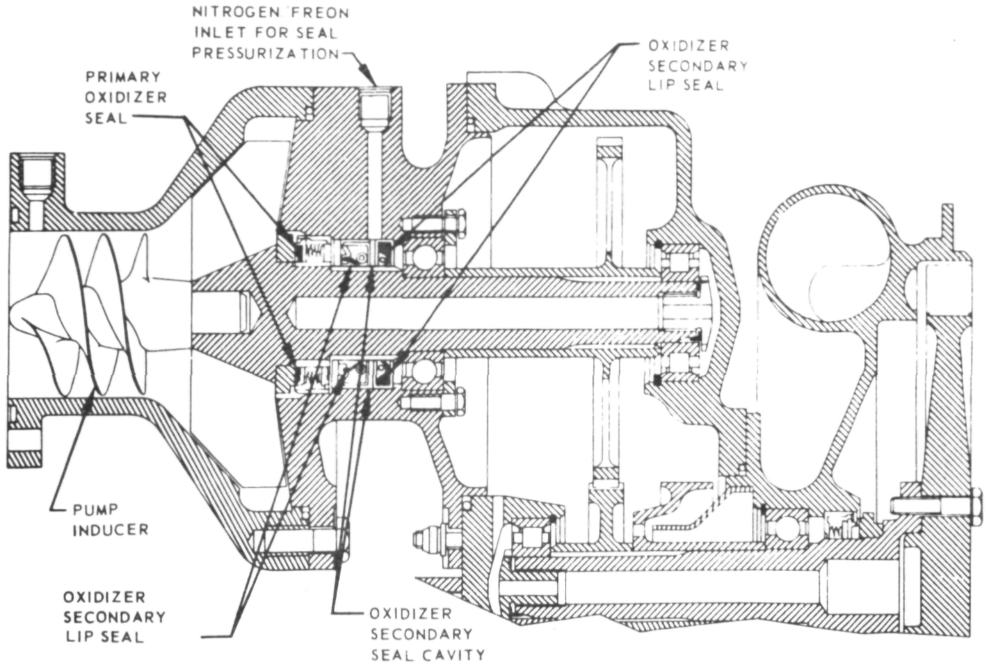
2.2.3.1 Solid-Propellant Starter. A single, dual, or triple solid-propellant starter assembly is used in the engine. The single starter assembly consists of an igniter assembly, a start can, and a bellows. The igniter assembly contains two pyrotechnic squibs and a smokeless powder booster charge. The start can holds a solid-propellant grain that consists of a cylinder of double-base solid propellant with a neutral-burning internal star configuration. The start can has an exit port that is sealed by a rupture disk prior to engine firing. The disk ensures proper ignition of the solid grain before allowing gases into the gas generator and provides dual- and triple-start installations with protection from hot gases entering and igniting the unburned starter cans. It can be ruptured in the normal flow direction by a pressure of 600 psi, but 1625 psi is required to burst it in the reverse flow direction. The interconnecting bellows is a short, flexible metal line that serves to isolate the vibration of the gas generator turbopump assembly from the starter can and permits thermal expansion between them. The bellows assembly terminates in a V-band clamp that mates to the top of the gas generator injector.

The dual solid-propellant starter assembly consists of two igniter assemblies, two start cans, a Y-fitting, and a bellows. The description and operation of the igniter assembly, start cans, and bellows are the same as for the single starter assembly. The two start cans are coupled together at a welded Y-fitting, which in turn is welded to the bellows. The triple solid-propellant starter uses a three-start-can configuration with three grains and igniters.

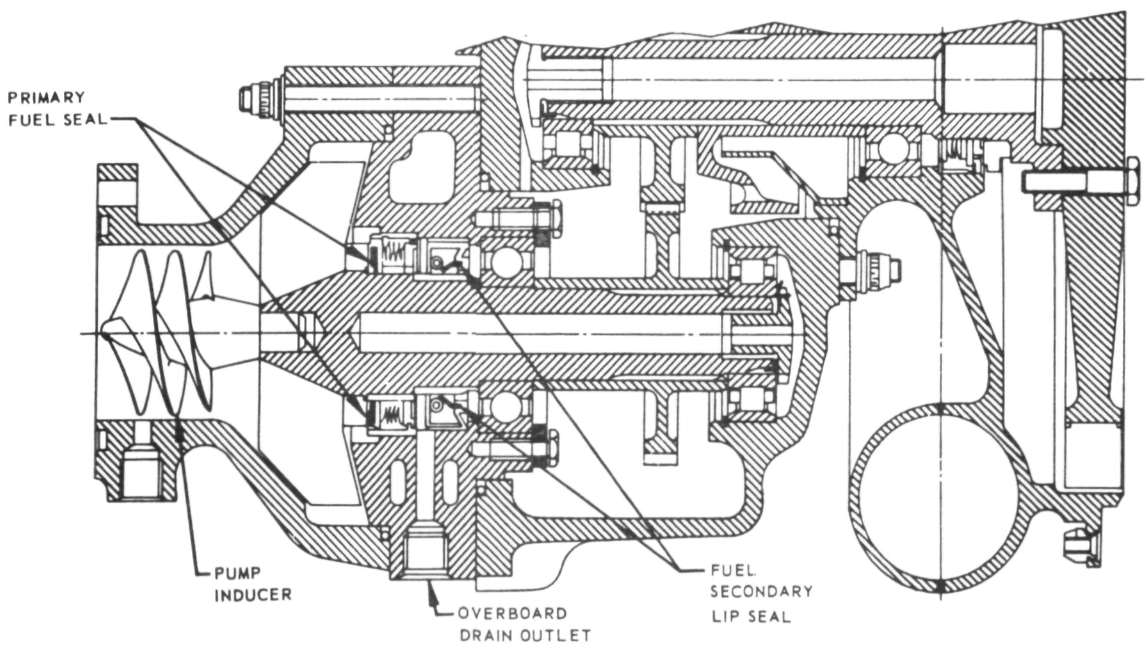
2.2.3.2 Turbopump Assembly. The propellant feed turbopump assembly consists of a single-stage impulse-type turbine, an oxidizer pump and a fuel pump that are gear-coupled to the turbine shaft, and a gear housing that serves as the assembly frame for the three major components. To minimize the possibility of an explosive mixture occurring in the gear case, the fuel pump shaft is sealed from the gear case with a primary sliding ring-type seal and a single-lip seal; the oxidizer pump is sealed with a primary seal and a double-lip seal. To provide maximum protection against oxidizer leaking into the gear case, this double-lip seal is pressurized with low-pressure nitrogen gas that forces any leakage past the primary seal to flow out an overboard drain line. Integral steel impellers and shafts are used in both the fuel and the oxidizer pumps. Aluminum inducers, which reduce the required pump inlet pressure for the pumps, are splined to the forward end of the pump impellers.

Turbine. The turbine is a single-stage impulse type designed to operate at 24,800 rpm. It is driven by hot gases from a bipropellant gas generator, whose spent gases are exhausted overboard through a turbine exhaust duct. The turbine inlet manifold incorporates the gas generator chamber and adapts to the gas generator injector. Turbine wheel rotation is initiated by the solid-propellant starter grain, which spins it up to approximately one-half the normal rated speed until liquid propellant combustion in the gas generator increases and maintains the turbine speed. The turbine gear case, which serves as a mount for the turbine and pumps, contains 150 cc of MIL-L-7808D turbine oil for lubrication of the gears and bearings. Actual lubrication is supplied by a mist produced from a slinger ring, mounted on the turbine drive shaft, which stirs up the oil.

Propellant Pumps. Centrifugal propellant pumps (Fig. 2-21) are used. They incorporate circular casings and straight vane impellers, with the flow terminating into a venturi-type diffuser. The diffusers are designed to cavitate, giving the pumps the desirable characteristic of flow rate in direct proportion to speed. This speed/flow control is independent of suction pressure changes over a wide range and is within the operating range of the pumps. Although not as efficient as pumps of conventional



OXIDIZER PUMP



FUEL PUMP

Fig. 2-21 Oxidizer and Fuel Pumps

design that do not provide ratio control, the pump eliminates the need for complex devices for controlling propellant mixture ratio. Consequently, increased reliability and reduced weight are achieved with only a small reduction of pump efficiency.

The pump inducers reduce the minimum allowable pump inlet pressures to approximately 12 psia for the fuel pump and 8 psia for the oxidizer pump. The design speeds for the fuel and oxidizer pumps are 25,389 and 14,410 rpm, respectively. The average pump flows are 15.27 lb/sec for fuel and 39.32 lb/sec for oxidizer. The oxidizer pump gas generator feed line is filtered to 250 microns; the fuel is filtered to 625 microns. The gas generator oxidizer flow passages are small in comparison to those for the fuel in the cavitating venturi and the injector orifices, hence the need for smaller particle restriction. Tangential takeoff ports on the fuel pump supply gas generator fuel flow through a venturi, actuation pressure for the gas generator bipropellant valve and fuel valve, main fuel flow to the thrust chamber through the fuel valve, and integrated hydraulic power package operating pressure. The tangential takeoff ports of the oxidizer pump supply gas generator oxidizer flow through a venturi and main oxidizer flow to thrust chamber injector through the oxidizer valve and thrust chamber coolant passages.

2.2.3.3 Gas Generator Assembly. The gas generator assembly sustains turbine operation after the solid-propellant starter charge burns out. It consists of a small combustion chamber, a bipropellant valve, a solenoid valve, and a pair of cavitating venturis for regulating the flow of oxidizer and fuel to the generator. The gas generator chamber is an integral part of the turbine inlet manifold.

Cavitating Venturis. The cavitating venturis, which provide turbine speed control, are made of stainless steel for corrosion and erosion resistance and are machined and polished to a smooth finish for high efficiency. They have a short converging section and a relatively long diverging section. The throat diameters vary, but are approximately 0.027 in. for oxidizer and 0.104 in. for fuel.

Bipropellant Valve. The gas generator bipropellant valve (Fig. 2-22) controls the flow of fuel and oxidizer to the gas generator. The pintle-type valve assembly consists of two poppet valves that are opened by a hydraulic actuator located between the valve bodies. The actuator stem is coupled to the two valve stems by a rigid bar so that the actuator stem and the valve stems move axially in unison when the valve assembly is opened or closed. The bipropellant valve is pilot-controlled by a solenoid valve that is integrally mounted on the bipropellant valve. The solenoid valve is closed at the same instant that the solid-propellant starter is ignited. When the solenoid valve is closed, fuel pressure can no longer pass through the actuator cavity of the gas generator bipropellant valve and return to the suction inlet of the pump. Therefore, as fuel pressure builds up under an actuation piston in the cavity, the bipropellant valve is forced open. Propellant flow then commences and the cavitating venturis begin regulating the flow. The liquid propellants enter the generator combustion chamber and ignite because of hypergolic reaction and solid-propellant charge combustion products.

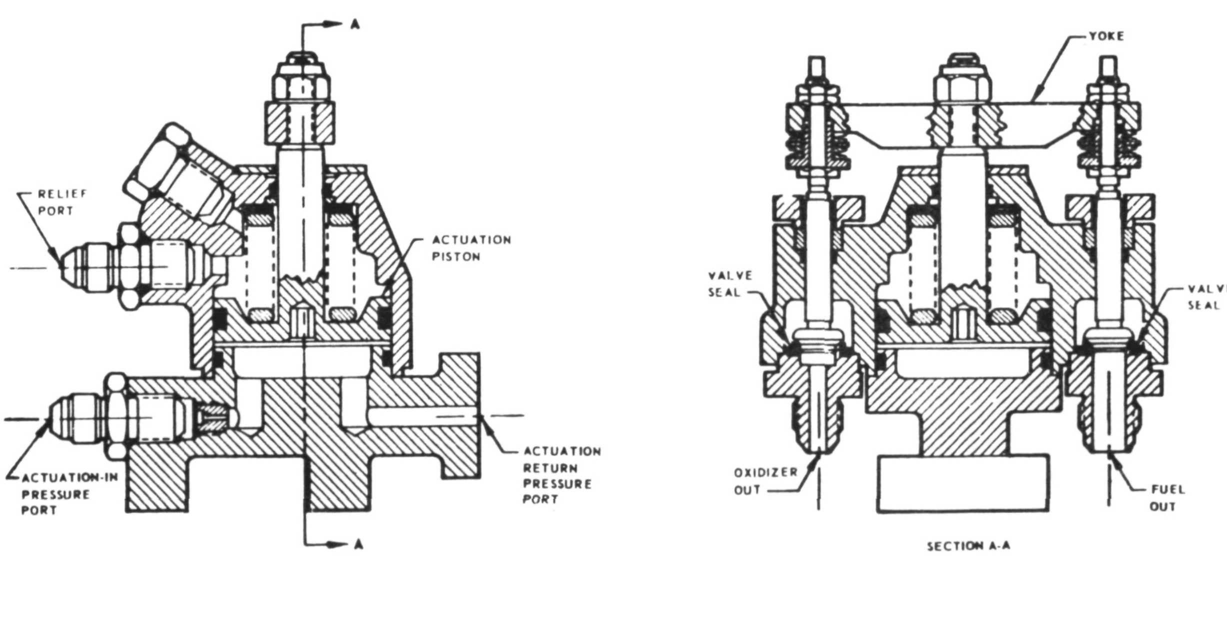


Fig. 2-22 Gas Generator Bipropellant Valve

Combustion Chamber. The gas generator is made of stainless steel and uses a fuel-rich mixture for cooling the combustion chamber. It operates at a rated mixture ratio of 0.150 (oxidizer) to 1 (fuel), which is highly fuel-rich compared to the engine thrust chamber mixture ratio of 2.81 (oxidizer) to 1 (fuel). Gas from the solid-propellant starting charge enters through an axial orifice at the injector end of the chamber and exits at the opposite end. The injector head, which is welded to the combustion chamber, has four primary oxidizer injector orifices and eight primary fuel injector orifices operating in a two-fuel-on-one oxidizer pattern. In addition, eight sets of one-on-one fuel doublets, located on the outer periphery of the injector plate, spray on and cool the chamber walls. The nominal gas generator chamber pressure is 485 psia, and the gas temperature is approximately 1400°F.

Gimbal Power Bleed. A fuel takeoff on the fuel pump diverts approximately 6.75 gallons of fuel per minute at approximately 1060 psia to drive a positive-displacement motor-pump assembly. The resultant hydraulic pressure operates electrically controlled actuating cylinders that gimbal the engine in its mount. The auxiliary motor-pump produces no noticeable effect on the main engine operation except for consumption of a small amount of gas generator propellant. The increase in flow rate for powering the auxiliary motor-pump is computed at 0.035 lb/sec.

2.2.3.4 Thrust Chamber and Propellant Valve Assemblies. The propellants unite in the engine thrust chamber to ignite and produce the required thrust. During the time that the gas generator and turbopumps are coming up to operating pressures and speeds, oxidizer is flowing through the feed lines to the thrust chamber. Although the operation of the turbine pump assembly has been described as if it were an isolated unit, the thrust chamber and turbine pump are closely related, both physically and operationally. Since the oxidizer is used as a thrust chamber regenerative coolant, the oxidizer aspects of the thrust chamber are described first. Description of fuel system functions related to the thrust chamber is deferred until that part of the sequence when the thrust chamber cooling passages are bled in and oxidizer is flowing overboard through the thrust chamber injector.

Thrust Chamber Construction. The thrust chamber is made of 6061 aluminum alloy and is fabricated in three sections: a combustion chamber, a nozzle throat, and a divergent nozzle section (13.3 to 1 expansion ratio). Each section is machined with integral cooling passages drilled through the walls of the combustion chamber and nozzle section. These passages are drilled in the combustion chamber in an axial direction around the periphery. The holes in the throat section are skewed to form a hyperboloid, and the divergent nozzle section contains holes drilled to conform to the contoured configuration. After machining, these sections are heliarc-welded together to make an integral assembly. This integral design provides a rugged structure; eliminates the need for external joints; and provides a smooth interior that reduces turbulence, promotes high heat-transfer rates, and decreases performance losses. The combustion chamber, throat area, and a small portion of the divergent nozzle section are coated with Keystone Aluminite Hardkote. At the 13.3 to 1 area ratio point, a radiation-cooled titanium nozzle extension is bolted to the cooled section of the nozzle. The titanium nozzle extension is coated with an aluminum oxide refractory coating on the inside and a high-emissivity surface on the outside. The extension is reinforced with molybdenum bands and stringers. The thrust chamber has a nozzle throat area of 17.12 sq in. and a nozzle extension exit area of 770.7 sq in. for the nominal expansion ratio of 45 to 1.

Oxidizer Valve. The oxidizer valve (Fig. 2-23) is a piston-actuated, spring-loaded, normally closed valve in which the poppet is also the actuating piston. Turbopump pressure of the oxidizer on the face of the poppet forces the poppet open against the spring. After opening at approximately 130 psig, oxidizer flows over the face of the piston and out through passages around the piston and spring cavity. On shutdown, the decaying oxidizer pump pressure allows the valve to close by spring action.

Oxidizer Fast-Shutdown Valve. An oxidizer valve fast-shutdown system is installed as standard equipment on dual- and triple-burn engines. This system, used at the termination of first burn, effects a fast engine shutdown and prevents an afterflow of oxidizer through the thrust chamber. Fast closing of the oxidizer valve is accomplished by the action of high-pressure gas applied to the back of the oxidizer valve actuating piston. The high-pressure gas is stored in a small cylinder mounted on the thrust chamber

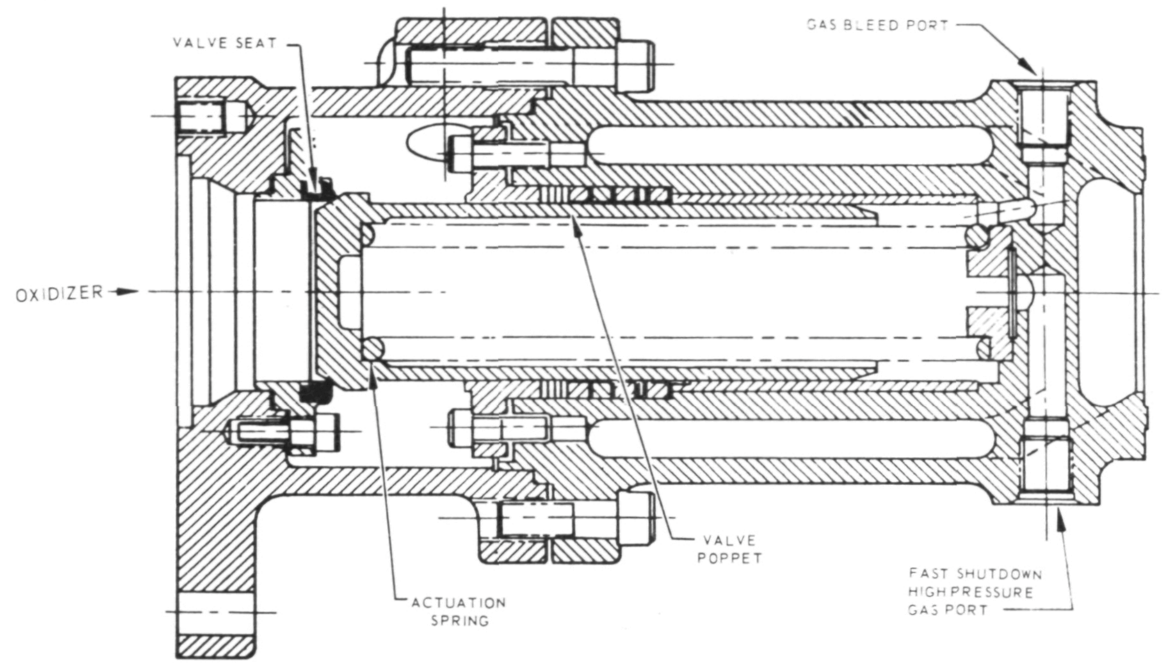


Fig. 2-23 Oxidizer Valve

and connected to the oxidizer valve through a normally closed, explosive-actuated Conax valve. When the valve is fired by the command signal to shut down the engine, the high-pressure gas enters the back side of the oxidizer valve piston through an inlet port and forces the oxidizer valve to close against the flow pressure. To enable the valve to open for the second start at the same pressure as the first start, the gas on the back side of the piston is allowed to vent through a small bleed orifice in the vent port. The fast oxidizer shutdown decreases postflow to 12 lb nominal versus a nominal postflow of 36 lb for a shutdown under spring force only.

Oxidizer Frangible Disk. An oxidizer frangible disk located downstream from the oxidizer valve serves a dual purpose: it prevents leakage into the thrust chamber cooling passages should the oxidizer valve leak liquid past its seat and provides the only gasket between the oxidizer valve and the thrust chamber inlet flange. The disk is constructed of two thin pieces of aluminum cemented together. The downstream side is notched with a predetermined rupture pattern; the upstream side is plain. Rupture pressure is approximately 180 psi, at which a trapdoor in the disk bursts and folds flat against the side of the oxidizer inlet manifold. Oxidizer can then enter the thrust chamber cooling passages.

Thrust Chamber Cooling. The oxidizer (IRFNA) enters the thrust chamber cooling passages through a manifold located slightly aft of the nozzle throat. The flow is divided and equalized in this ring by baffles. The oxidizer flows aft through the divergent nozzle section and then forward through the full length of the nozzle, manifold, chamber, and into the injector manifold.

Thrust Chamber Injector. The engine thrust chamber incorporates a triplet injector; that is, an injector in which a three-orifice pattern is repeated. Each triplet sprays two fuel jets on one oxidizer jet. A ring of fuel doublets is spaced around the perimeter of the injector for film cooling. The injector body is machined from aluminum alloy forgings and is heliarc-welded into the thrust chamber assembly. The injector has a cavity into which the fuel valve fits, allowing the fuel valve poppet to be located close to the fuel injector orifices. Free volumes are thus held small, so that when the fuel diaphragm ruptures, fuel injection starts with minimum delay and with minimum "hammer" as the free space fills with propellants. A similar arrangement is not needed for the oxidizer, since oxidizer flow is accelerated gradually, being injected for an interval prior to fuel valve opening. An additional advantage of locating the fuel valve close to the injector is that small, free volumes of fuel minimize the tailoff impulse and speed thrust termination after shutdown is initiated.

Oxidizer Manifold-Pressure Switch. The oxidizer manifold-pressure switch (OMPS) is used to detect the presence of oxidizer in the injector manifold of sufficient pressure to have filled all the cooling passages. At this point, a signal is sent to the fuel valve and pilot-operated solenoid valve to start the fuel flow to the injector. The oxidizer manifold pressure switch operates through the use of an expanding bellows and a leaf spring. Pressure entering the sensing port expands the bellows against the leaf spring, causing it to deflect toward a microswitch. Operation pressure of the OMPS is 33 psig make and break. On later engines, 8096-475300-35 and up, two OMPS are used for redundancy.

Thrust Chamber Pressure Switch. The thrust chamber pressure switch is used only on engines through the 8096-475300-31 configuration. This switch actuates when thrust chamber pressure reaches 391 psig; in conjunction with the engine relay box, it removes the OMPS from the shutdown circuit.

Fuel Frangible Disk. The fuel frangible disk is a fuel-line seal, similar to the oxidizer frangible disk, that prevents propellant leakage into the fuel valve. Such leakage, exiting through the fuel valve into the thrust chamber nozzle, could cause damage to the vehicle booster during the period prior to vehicle-booster separation. During the engine start sequence, the frangible disk ruptures when the fuel pressure builds up to 525 psig; the fuel valve opens shortly thereafter. Since satellite-booster separation has already occurred, the frangible disk is not required for second or subsequent burn operations. The fuel frangible disk is mounted in a holder and attached to the inlet of the fuel valve. A fuel inlet screen is located just downstream of the disk (inside the fuel valve) to trap any particles of the ruptured disk that might plug fuel orifices or hold the fuel valve off its seat.

Fuel Valve and Pilot-Operated Solenoid Valve. The fuel valve (Fig. 2-24) is a spring-loaded pressure-actuated, poppet-type valve attached to the aft portion of the engine thrust chamber. The fuel valve is normally held in the closed position by spring pressure and is opened by fuel pressure applied to an actuating piston that is an integral part of the fuel valve. The pilot-operated solenoid valve (POSV) controls the application of fuel pressure to the fuel actuation piston and thus controls the opening and closing of the fuel valve. When the POSV (Fig. 2-25) is actuated upon receipt of the oxidizer manifold-pressure switch(s) signal, fuel is prevented from flowing out of the fuel valve actuation cavity. This causes actuation pressure to build up and force the fuel valve open. At engine shutdown, the POSV is deenergized by removal of power from the engine. Deactuation of the POSV results in a rapid decay of fuel valve actuation pressure, and the fuel valve closes under spring pressure.

2.2.3.5 Engine Mount and Gimbal Assembly. The engine mount (Fig. 2-26) consists of a tubular steel frame that is attached to the satellite vehicle. The thrust chamber assembly is attached to the engine mount on a mutually perpendicular, two-axis gimbal system to provide for vehicle pitch and yaw control. The gimbal mount system provides for 5 deg of thrust chamber deflection in a square pattern. However, when it is installed in a vehicle, the thrust chamber deflection is limited to approximately 3 deg. A ring

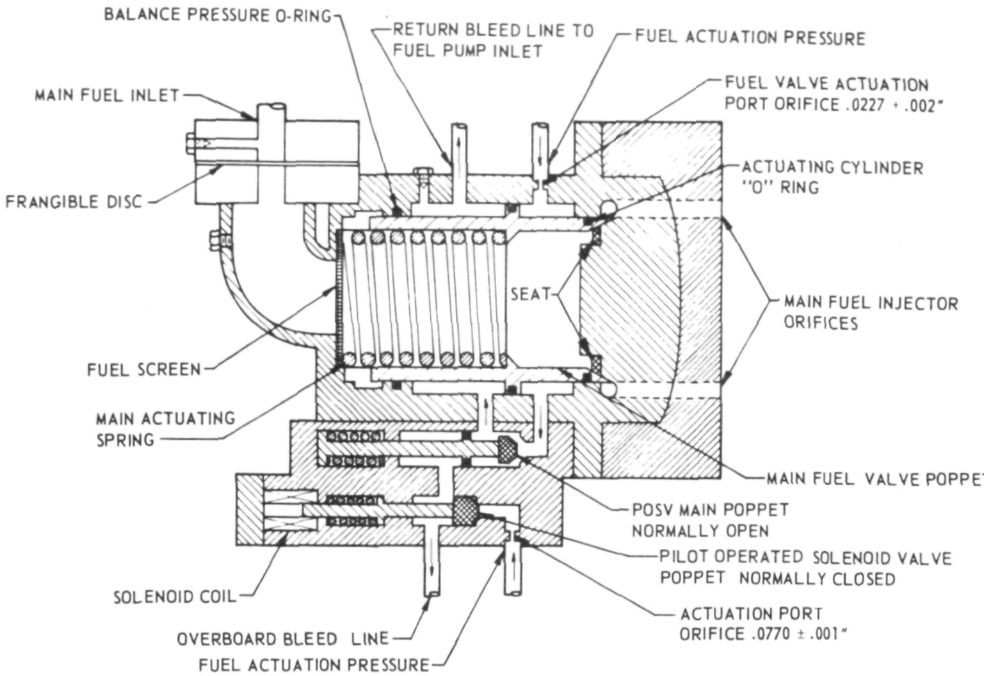


Fig. 2-24 Fuel Valve

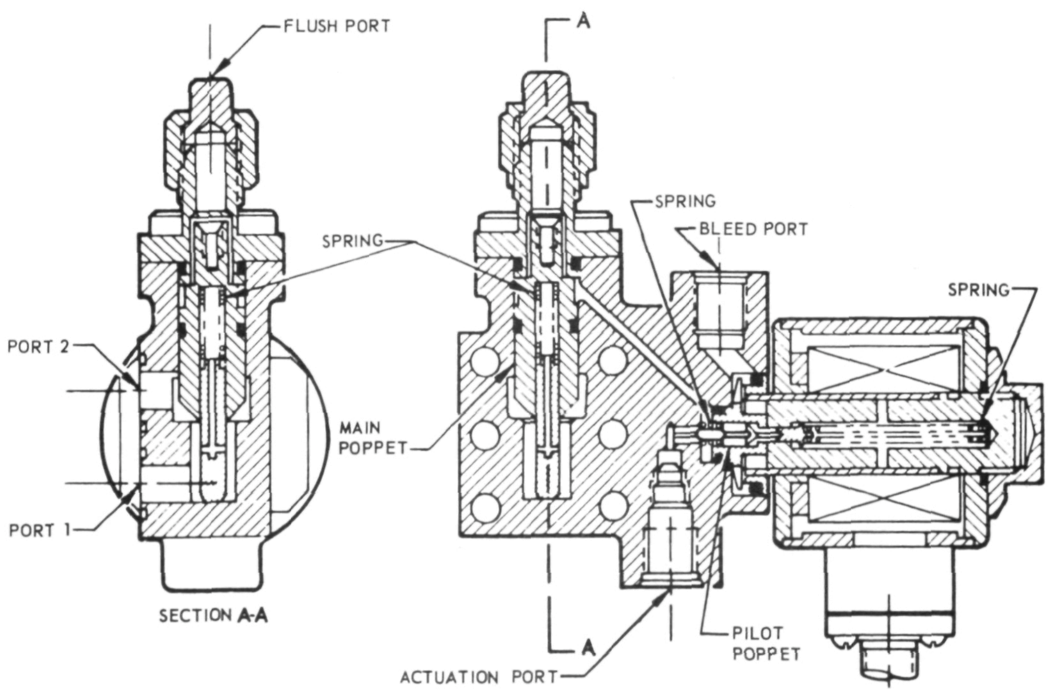


Fig. 2-25 Pilot-Operated Solenoid Valve

containing four equally spaced bearings surrounds the thrust chamber forward of the injector. Pivots for these bearings are provided by close-tolerance pins, two of which are threaded and locked into pads welded to the forward face of the injector. Rotation of these two pins provides deflection in the lateral plane. In a similar manner, two additional pins are threaded through the engine mount structure to provide pivots for the other two bearings that permit deflection in the perpendicular plane. Thrust is thus transmitted through each pair of bearings to the mount structure.

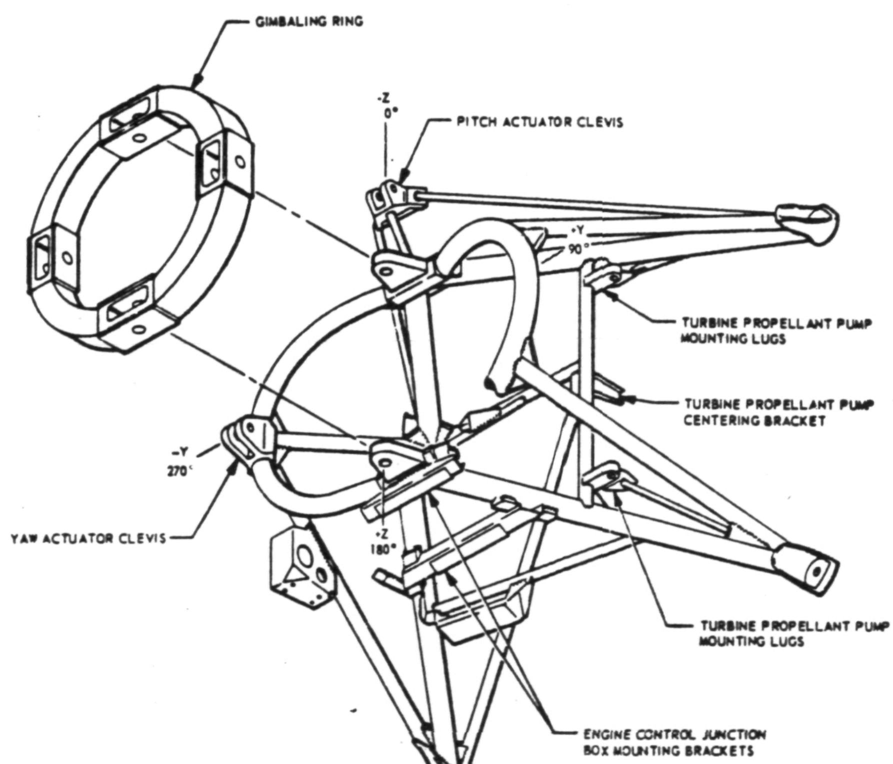


Fig. 2-26 Engine Mount and Gimbal Ring Assembly

2.2.3.6 Electrical Control Equipment. There are two areas to consider in discussing the engine electrical equipment: the engine pyrotechnic circuits and engine control circuits. Both of these circuit groups can be traced by reference to Fig. 2-27.

Engine Pyrotechnics. The number of types of pyrotechnic devices installed depends on whether the engine is configured for single, dual or triple start. Single-start engine installations require only one starter igniter assembly (containing two igniter cartridges) and do not require oxidizer valve fast shutdown, since there is no need to control oxidizer postflow. Dual- and triple-start engine installations have two or three starter igniter assemblies and incorporate the oxidizer valve fast shutdown. The signals to activate the engine pyrotechnics originate in the vehicle sequence timer; power from the vehicle pyro bus is used. All engine pyro event signals are routed through an aft safe-arm box safety plug circuit to provide the safe and arm functions required for personnel and equipment safety.

Engine Control Circuits. Engine electrical controls for engines through 8096-475300-31 incorporate a relay box mounted on the engine thrust mount. For engines 8096-475300-33 and up, the relay box has been eliminated and its functions performed by the vehicle aft safe-arm box.

Figure 2-27 shows the electrical control schematic when a relay box is used. Whenever main bus power is applied to the vehicle, unregulated, unswitched 28 vdc is received at the engine relay box through the engine-to-vehicle interface plug, P4000. There are two receptacles on the engine relay box: J4101 and J4116. A few of the individual pin receptacles in J4101 receive vehicle power and command inputs from P4000 and its attached harness: the remaining receptacles are used for solenoid valve energizing voltages and pressure switch control through the engine control harness. J4116 is used exclusively for transmission of instrumentation signals from the engine control system to the aft instrumentation box. The output signals from J4116 are used to indicate the position of relays, solenoids, and pressure switches during engine operation.

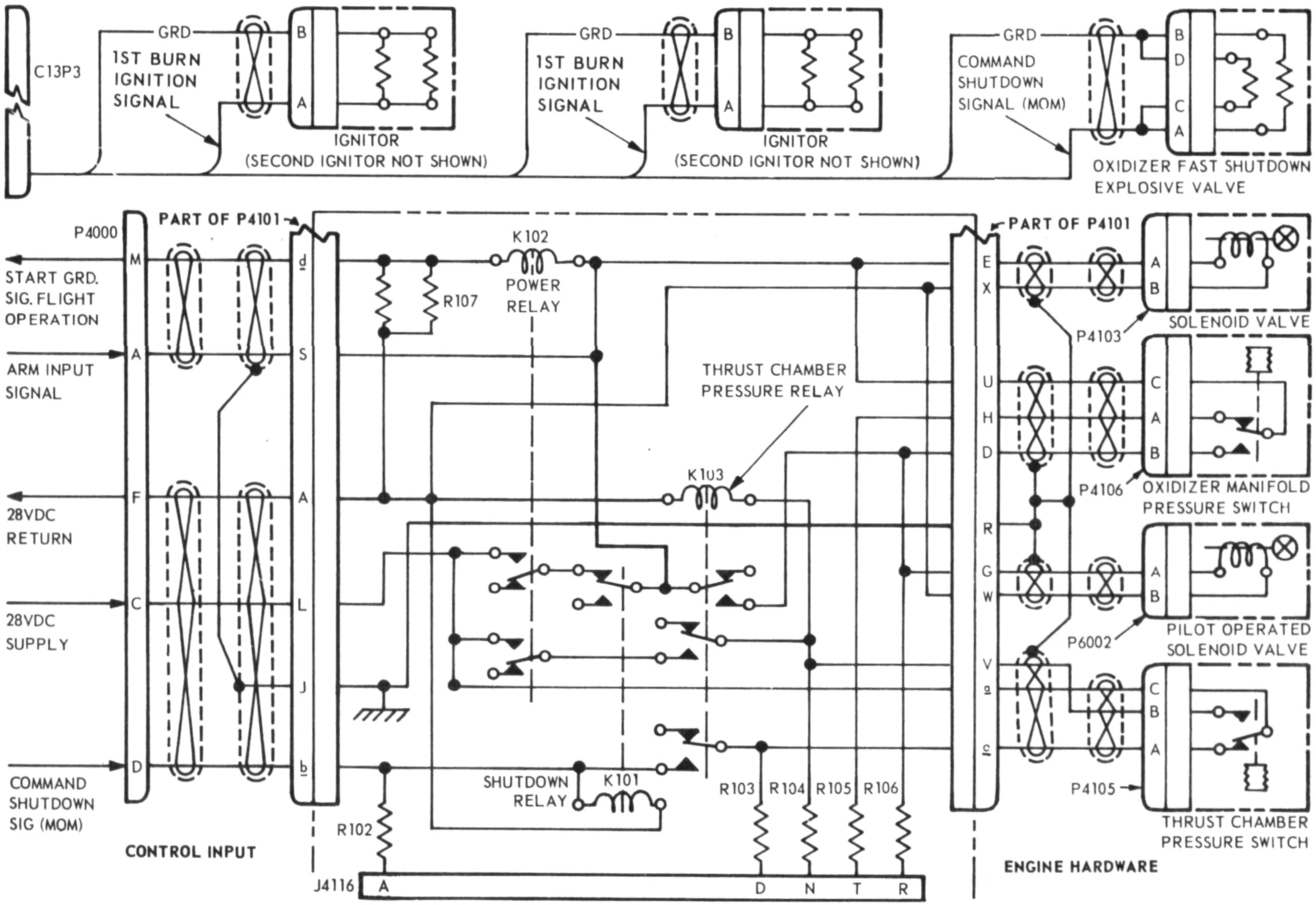


Fig. 2-27 Engine Electrical Control Schematic

Engine start is initiated by applying 28 vdc at the arm input signal pin of the engine relay box, which energizes the power relay K102. K102 holds itself in by connecting, through a pair of its own contacts and through the normally closed contacts of the shutdown relay, K101 to the 28 vdc unregulated, unswitched source. As K102 is energized, power is directed to close the gas generator solenoid valve; simultaneously, the sequence timer sends a pyro signal to ignite the solid-propellant starter. The turbine spins up and pressure begins to build up in the oxidizer feed lines, thrust chamber cooling passages, and finally in the oxidizer manifold. The oxidizer manifold pressure switch actuates and sends a signal to energize the pilot-operated solenoid valve and open the fuel valve. When fuel enters the thrust chamber and mixes with the oxidizer, it initiates combustion and builds up chamber pressure.

As thrust chamber pressure builds up, the thrust chamber pressure switch actuates (at 391 psig), which in turn energizes the thrust chamber relay K103. When K103 is energized, it also holds itself in by connecting its coil to the 28 vdc unswitched, unregulated supply through one pair of its contacts. K103 now supplies direct power to the pilot-operated solenoid valve, causing it to remain energized even if the oxidizer manifold pressure switch contacts chatter or open. The engine will shut down after sufficient burn time has elapsed if switched 28 vdc power is removed from the engine and K101 is energized simultaneously from an external command shutdown signal. The thrust chamber pressure switch (TCPS) may also shut down the engine if the switched 28 vdc is removed by relays in the aft safe-arm box. As a program-peculiar option, the engine circuitry can be configured so that a chamber pressure switch shutdown occurs when the thrust chamber pressure drops below 372 psig. In such an instance, the TCPS cycles back to its original position, and power is applied through contacts of the thrust chamber relay to the shutdown relay, removing all power from the solenoids.

The engine control circuitry is highly simplified when all sequencing is accomplished by a program-peculiar aft safe/arm J-box, as shown in Fig. 2-28.

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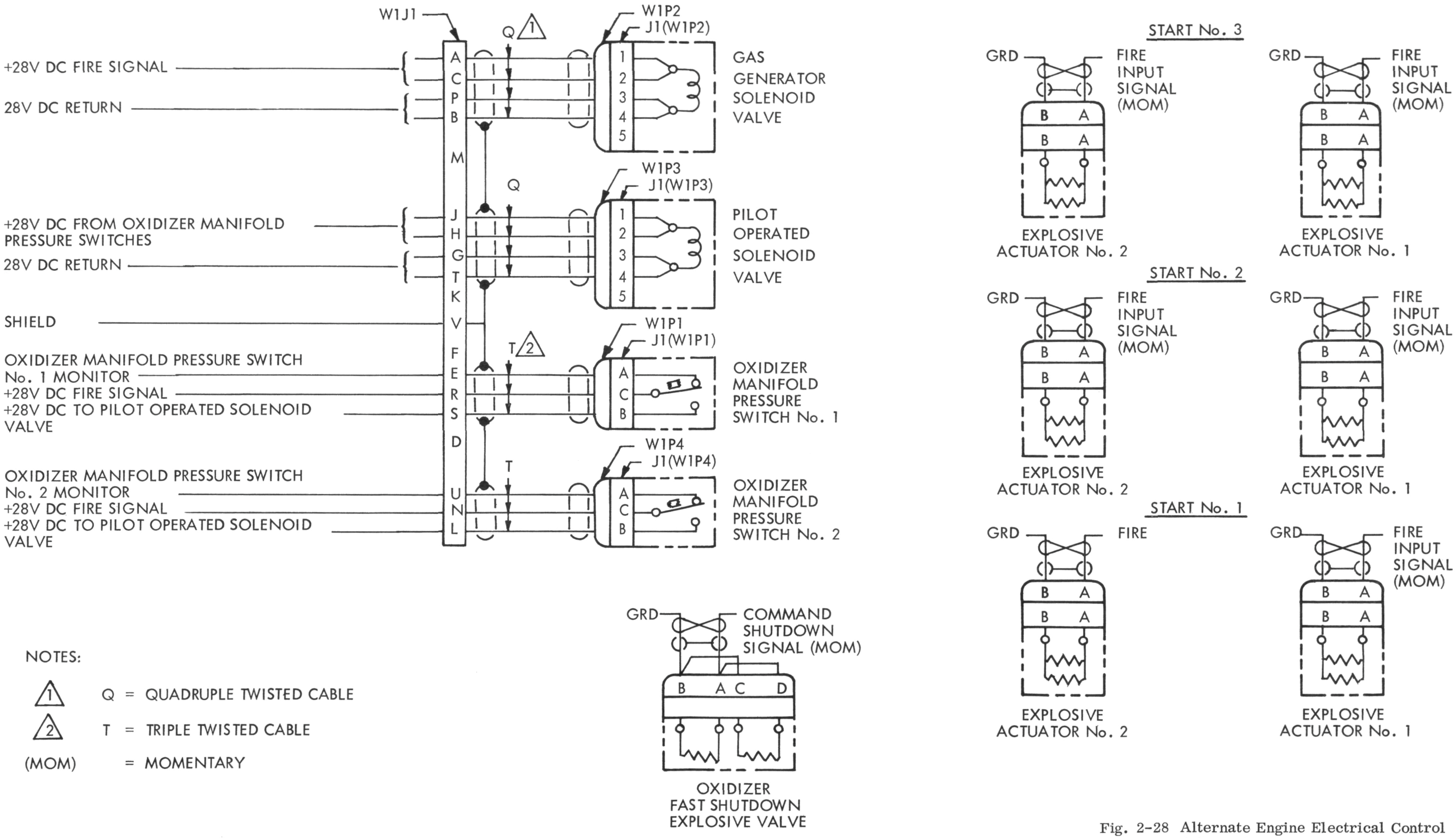

Fig. 2-28 Alternate Engine Electrical Control Schematic

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2.2.4 Engine Operational Sequence

2.2.4.1 General Start.* When propellants at suitable pump inlet pressures are supplied, engine operation begins when the vehicle sequence timer sends out two simultaneous 28-volt signals. One signal, from the main power bus, closes the gas generator solenoid valve (GGSV), blocking off the outlet passage from the actuating chamber of the gas generator bipropellant valve. The other signal, from the pyro power bus, fires the igniter cartridges of the turbine pump starter igniter assembly. Subsequent ignition of a solid-propellant grain starts the generation of hot gases, which are directed through the combustion chamber of the gas generator assembly to drive the turbine wheel of the turbine pump assembly. The turbine wheel drives the fuel and oxidizer pumps through a gear-coupled drive, causing the fuel and oxidizer discharge pressures to rise. The turbine starter grain has sufficient duration and power to drive the turbine to about half of nominal rated speed (12, 000 to 14, 000 rpm) for a period of about 1.3 sec.

Fuel pump output pressure builds up in the gas generator bipropellant valve actuating chamber, causing the actuating piston of the valve to open (at about 225 psi) the fuel and oxidizer poppets. The open poppets permit fuel and oxidizer to flow through their respective flow rate control venturis and into the gas generator injector. As propellants are injected into the generator chamber, they are ignited by hypergolic action, causing the turbine to accelerate toward its rated speed of 24, 800 rpm. The combined burning of the liquid and solid propellants produces a momentary turbine overspeed condition which exists until the solid-propellant grain burns out, at which time (about 1.3 sec after engine start) the turbine and pumps gradually coast down to rated speed and nominal pressures and flows. With the gas generator bipropellant valve open and propellants from the pumps being injected into the gas generator combustion chamber, the turbine pump assembly operates until a command shutdown signal is sent or until propellant exhaustion occurs.

*Engine propellant flow is diagrammed in Fig. 2-29.

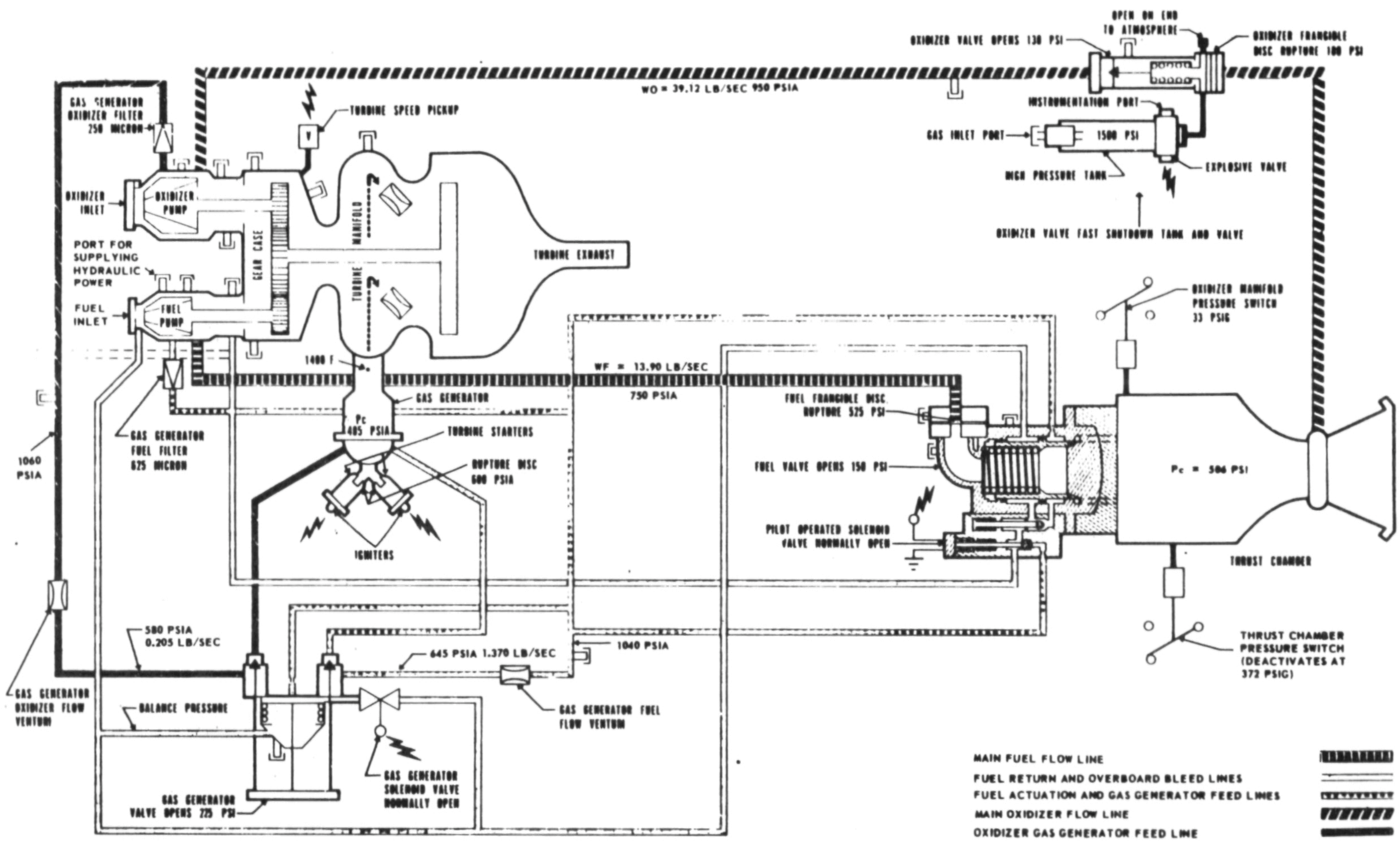


Fig. 2-29 Engine Propellant Flow Schematic

Slightly before the gas generator bipropellant valve opens, pressure from the oxidizer pump builds up to 130 psig in the propellant line, causing the main oxidizer valve to open by fluid pressure on a spring-loaded poppet. A frangible disk, immediately downstream from the main oxidizer valve, ruptures at 180 psig and permits oxidizer to flow into the cooling passages of the thrust chamber. The oxidizer flow enters an oxidizer manifold that surrounds and empties into the thrust chamber injector. When oxidizer pressure in the manifold reaches 33 psig, the oxidizer manifold pressure switch (OMPS) sends a signal to operate the pilot-operated solenoid valve (POSV). This sequencing assures that oxidizer is available in the cooling passages at ignition and is injected into the thrust chamber before the fuel, preventing a possible explosion or hard start.

When the OMPS senses 32 psig, the switch contacts close and 28 vdc is applied to close the POSV. The POSV and fuel valve operate together in much the same manner as the gas generator bipropellant valve and solenoid valve. That is, when the POSV closes, the outlet of the actuation chamber of the fuel valve is blocked, causing a pressure buildup that opens the piston-poppet valve assembly. As fuel pump pressure builds up to 525 psig, a frangible disk located at the inlet of the fuel valve ruptures; and fuel begins to flow across the open valve seat and the thrust chamber. The mixture ignites hypergolically and sustains combustion until the propellant source is exhausted or until a shutdown signal is applied to the electrical system.

The thrust chamber pressure switch (TCPS), sensing hot gas from the thrust chamber, actuates at 391 psig and connects 28 vdc to energize the thrust chamber relay. The thrust chamber relay, which is located in the engine relay box, connects 28 vdc directly to the POSV to permit the engine to operate even if the oxidizer manifold pressure switch should malfunction mechanically or electrically.

2.2.4.2 General Shutdown. A command shutdown signal is initiated by the vehicle velocity meter, or as a backup, the vehicle sequence timer. The shutdown signal from either source opens relays in both the aft safe-arm box and the engine relay box, when installed. Opening these relays removes the 28 vdc from the POSV and gas generator solenoid valve, causing both valves to return to their normally open positions.

The fuel valve and gas generator bipropellant valve actuation pressures decay, and both valves close. As soon as the fuel valve closes, the supply of propellants to the combustion chamber of the gas generator is shut off and the pressure decays. As the turbine wheel slows down, the oxidizer pump discharge pressure drops and allows the oxidizer valve to close.

2.2.4.3 Fast Shutdown (First Burn). On dual- and triple-burn vehicles, an oxidizer valve fast-shutdown system is installed as standard equipment to force the oxidizer valve to close early. The fast-shutdown system saves about 25 lb of oxidizer that would normally be dumped overboard through the cooling passages and injector of the thrust chamber. Engine first-burn shutdown impulse is also more predictable with this system. Operation of the fast-shutdown system is initiated when the engine shutdown signal is sent. A separate 28 vdc signal from the pyro bus is sent to the fast-shutdown explosive valve, which opens, releasing 1500 psig of stored nitrogen gas into the spring chamber of the oxidizer valve. The combination of normal spring forces and the nitrogen push the oxidizer valve poppet closed more quickly than the spring could by itself.

2.2.4.4 Restart. After the high-pressure nitrogen has vented through an orifice to ambient pressure, the engine is ready for restart. Because of venting time, the minimum coast period is 15 sec. The engine restart sequence parallels the initial sequence, except that the second or third solid-propellant starter is used and the fuel and oxidizer frangible disks, ruptured at first burn ignition, offer no resistance to propellant flow. These disks are not essential to restart operations, since their primary purpose is to protect the Agena booster from propellant that might leak through valve seats before engine ignition.

2.2.4.5 Second or Third Shutdown. Second- or third-burn shutdown is initiated by a signal from the vehicle velocity meter or sequence timer, and events occur in the same manner as the first-burn shutdown, except that the oxidizer fast-shutdown system does not function. The last-burn shutdown signal may also be used to open an oxidizer dump valve if this kit has been installed.

2.2.4.6 TCPS-Initiated Shutdown. An alternate method of engine shutdown that will provide a smooth positive shutdown sequence in the event of engine malfunction or propellant depletion may be selected. If the thrust chamber pressure decays to 372 psig, the thrust chamber pressure switch deactuates and, through contacts of the thrust chamber relay, energizes the shutdown relay, which in turn removes power from both control solenoids. Thrust chamber pressure and turbine speed then decay, causing shutdown. A TCPS shutdown can occur only if the control relays in the aft safe-arm box removes the switched 28 vdc (defined as power that can be controlled by sequence timer or relays) from the engine relay box. Unless optionally configured, a TCPS shutdown cannot occur because the switched 28 vdc is not removed until shutdown is commanded. In a propellant depletion shutdown (soft shutdown) without TCPS control, the engine stops as a result of ingested gas entering the pumps, which interrupts propellant flow to the gas generator and thrust chamber. As ingested gas begins to interfere with gas generator operation and actuation as well as feed pressures decay, closure of the gas generator bipropellant and thrust chamber fuel valves terminates engine operation.

2.3 ELECTRICAL POWER SYSTEM

The Agena electrical power system is an unregulated 28-volt system in which two Type IVB batteries serve as independent power sources: one supplies power to the main electrical bus; the other, power for operation of the vehicle pyrotechnics. The pyro battery is also used as a backup for supplying vehicle main power in the event of a low-charge condition in the main battery.

Payload support functions, such as pyro control for spacecraft separation, ascent electrical power, ascent telemetry and spacecraft control, can be supported from the Agena electrical system through the payload pyro and control J-box. Since payload support requirements vary, this J-box can be changed to accommodate individual spacecraft needs. Through the J-box, the spacecraft can interface with either analog or telltale monitors on the Agena telemetry, pyrotechnic power control relays, and electrical power buses. In addition, prelaunch checkout, monitoring, and control via the Agena unbilical can be accommodated (Fig. 2-30).

The electrical system circuit designs are compatible with anticipated program requirements. The equipment physical features include welded-magnesium enclosures for high strength-to-weight ratios and low manufacturing cost and the use of plug-in printed circuit modules for mounting the individual box components. This latter feature is conducive to fast assembly time and facilitates the interchangeability of parts.

2.3.1 Power Sources

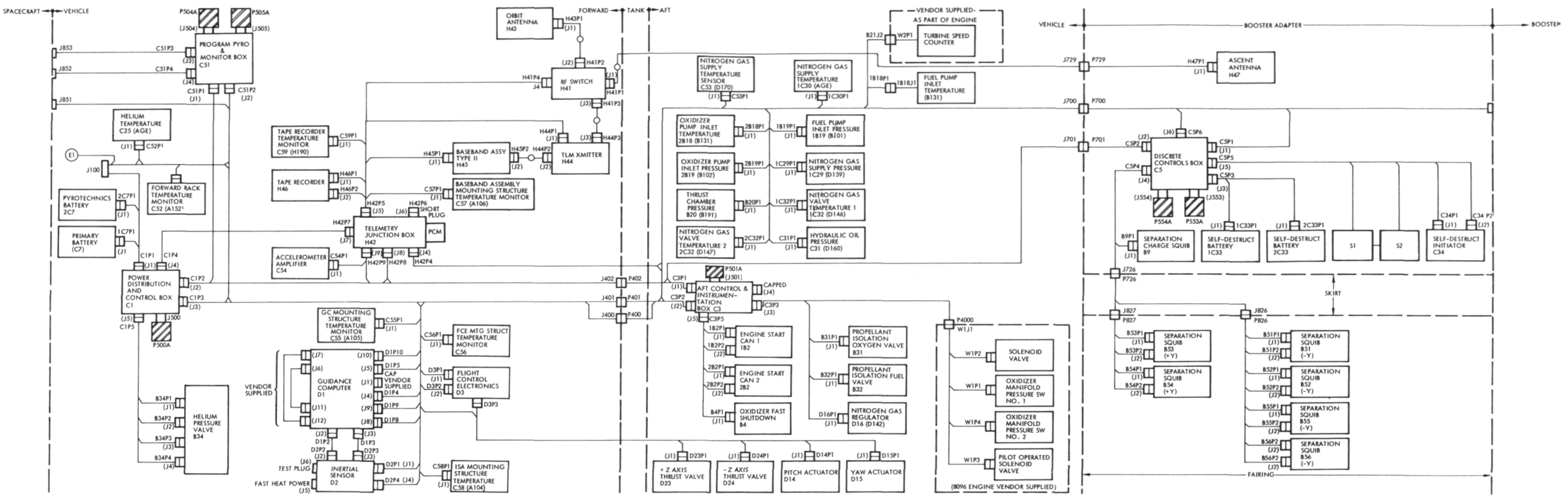
2.3.1.1 Batteries. The two Type IVB batteries are connected in parallel to a main power bus; however, the pyrotechnic battery is diode-isolated. In this way, the pyrotechnic supply is used to back up, but does not impose pyrotechnic loads on the main supply. The main supply and pyro supply are connected to the main distribution and pyro power distribution systems through a double-pole power control switch located on the vehicle and operated by ground control. An external source of power supplies the Agena loads while the batteries are installed and the Agena power control switch is open. A two-wire system is employed to distribute the unregulated battery power throughout the vehicle, with the return wire referenced to structure potential at a single point near the batteries.

The diode-isolated battery supply used to actuate pyro devices precludes the occurence of negative spikes on the main vehicle bus. A nonlatching, double-throw relay prevents inadvertent detonation of pyro devices by induced or electrostatic potential; this is accomplished by maintaining both sides of the bridgewire at ground potential through the back contacts of the double-throw relay before pyro initiation.

Primary batteries of varying capacities as shown below may be selected; however, two Type IVB batteries provide enough power for the basic ascent Agena.

PRIMARY BATTERY CHARACTERISTICS

LMSC Battery	Cells	Nominal Voltage	Ampere-Hours	Mean Watt-Hours	Weight (lb)
Type IVB	18	27.5	16	440	17
Type VIA	17	26.0	45	1,170	26
Type IC	16	24.5	450	11,025	118
Type IK	16	24.5	475	11,637	128
Type 30	18	27.5	400	11,000	134



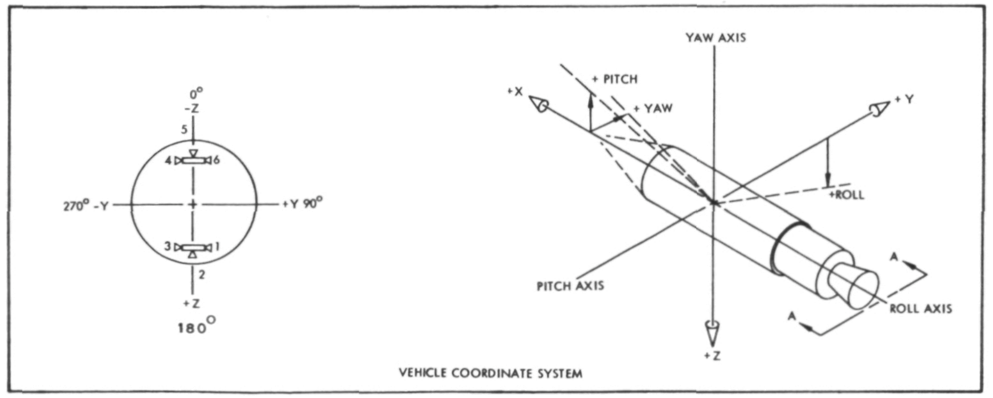


Fig. 2-30 Ascent Agena Interconnect Diagram

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A typical mission power summary is shown below. For larger battery capacity requirements, mounting kits that can accommodate up to three of the Type 1K or Type 30 batteries have been developed.

MISSION POWER SUMMARY

Item	Minutes	Watt-Hours
Prelaunch	8	42.5
Ascent	10	57.6
Coast	20	107.3
Wire Loss (2%)		4.1
Total Requirement		211.5
Battery Capability (Two Type IVB)		880.0
Surplus	127	668.5

2.3.1.2 Solar Arrays. Primary electrical power for the Agena is configured for a specific mission. To support long-term missions, solar arrays of rigid panels and extendible panels have been successfully flown. A flexible substrate, rollup solar array design is being developed that can deliver up to 3000 watts of power. Solar arrays usually charge nickel-cadmium secondary batteries; however, on some flights the arrays have been used to charge specialized silver-zinc primary batteries. Regulation of battery charging is provided by one of several flight-qualified charge controllers.

2.3.2 Electrical Control and Distribution Equipment

The principal items of electrical control and distribution components are described in the following paragraphs.

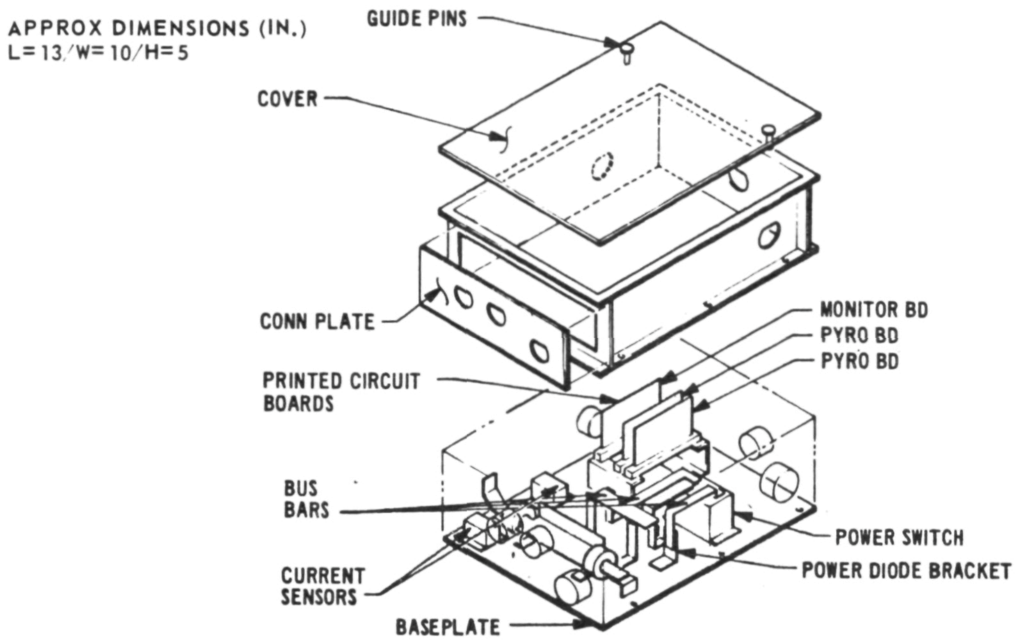
2.3.2.1 Power Distribution and Control Box. This unit (Fig. 2-31) consists essentially of bus bars, current sensors, and a power transfer switch for performing main power distribution, control, and monitoring functions, plus printed circuit modules for pyro control circuits and signal conditioning. The unit weighs approximately 10 lb and is located in the Agena forward section.

External and internal power of 22 to 29.5 vdc is switched and distributed on both the main power bus and the pyro power bus. A 35-amp diode isolates the main bus from the pyro bus under normal conditions but automatically passes pyro battery power to supply the vehicle loads when the main battery is approximately 0.8 volt lower than the pyro battery.

2.3.2.2 Telemetry Junction Box. This equipment (Fig. 2-32), located in the Agena forward section immediately adjacent to the Type 4 PCM telemeter unit, weighs approximately 5 lb. Vehicle data monitor signals are received by this box and commutated through a terminal junction network to the PCM channel inputs. Unused channels are shorted to telemetry ground. The box also routes and controls power, AGS, and GSE signals for controlling the UHF transmitter, tape recorder, baseband assembly unit, and PCM telemeter unit on and off. The AGS serial data are controlled through a latching relay to the tape recorder.

2.3.2.3 Program Pyro and Monitor Box. This unit (Fig. 2-33) is also located in the Agena forward section. It consists essentially of five printed circuit assemblies mounted within a deep-drawn enclosure and weighs approximately 5 lb.

The functions of this box are concerned with spacecraft separation and monitoring. Upon receipt of a command signal from the AGS guidance computer (GC), the box delivers current through fusistor-protected redundant circuits to the spacecraft separation bolts. The GC command is also routed to the signal-conditioning module (divider network) and sent to the PCM telemeter. A similar signal-conditioning circuit (not shown in Fig. 2-33) is provided for spacecraft separation monitoring.



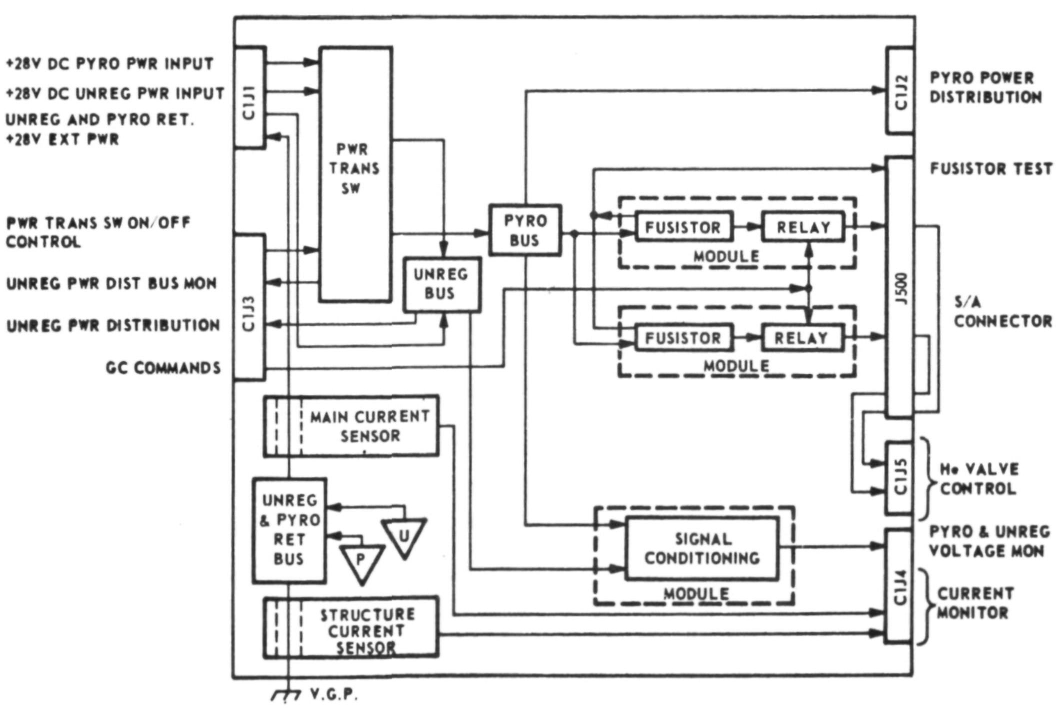


Fig. 2-31 Power Distribution and Control Box

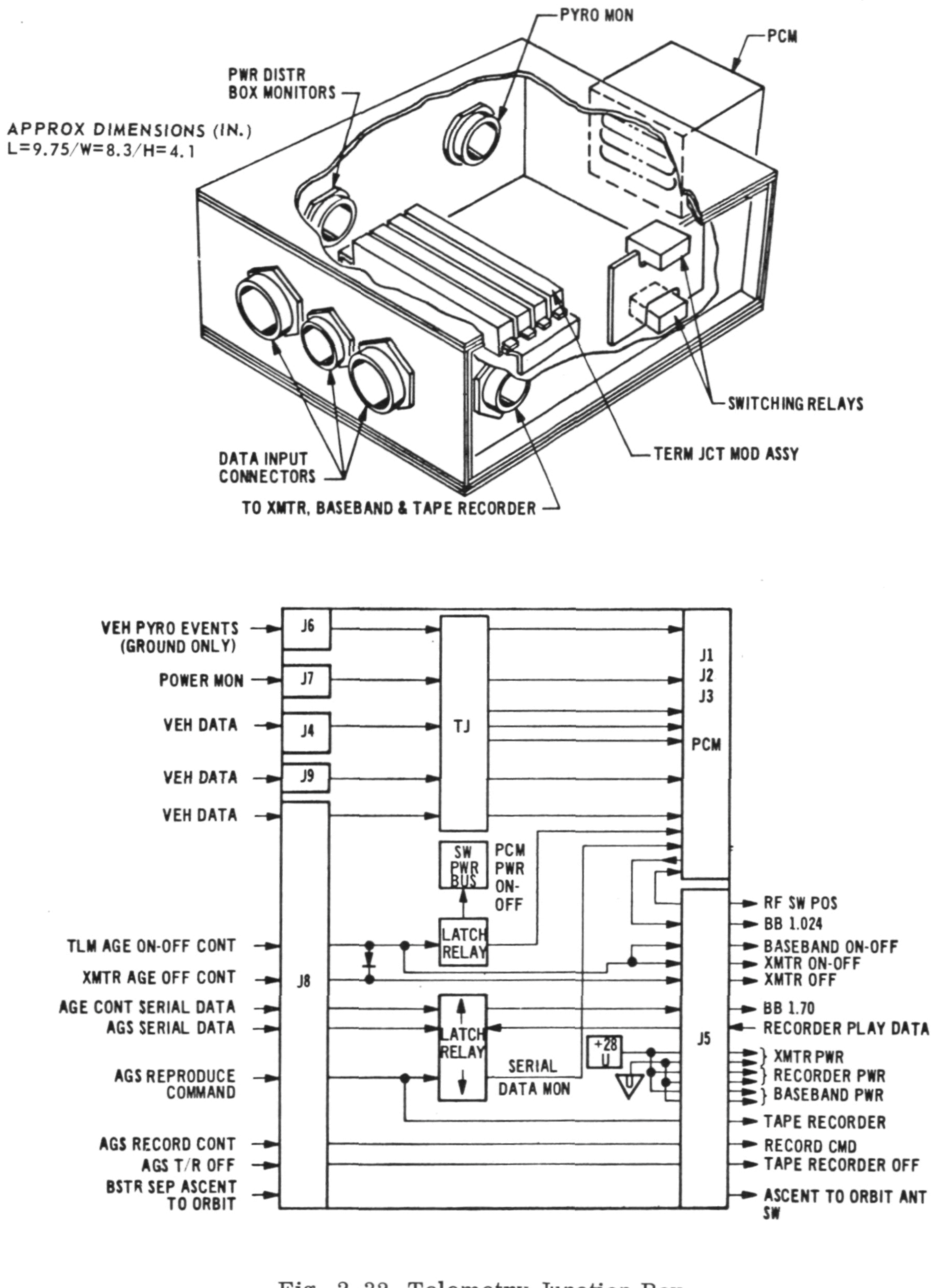


Fig. 2-32 Telemetry Junction Box

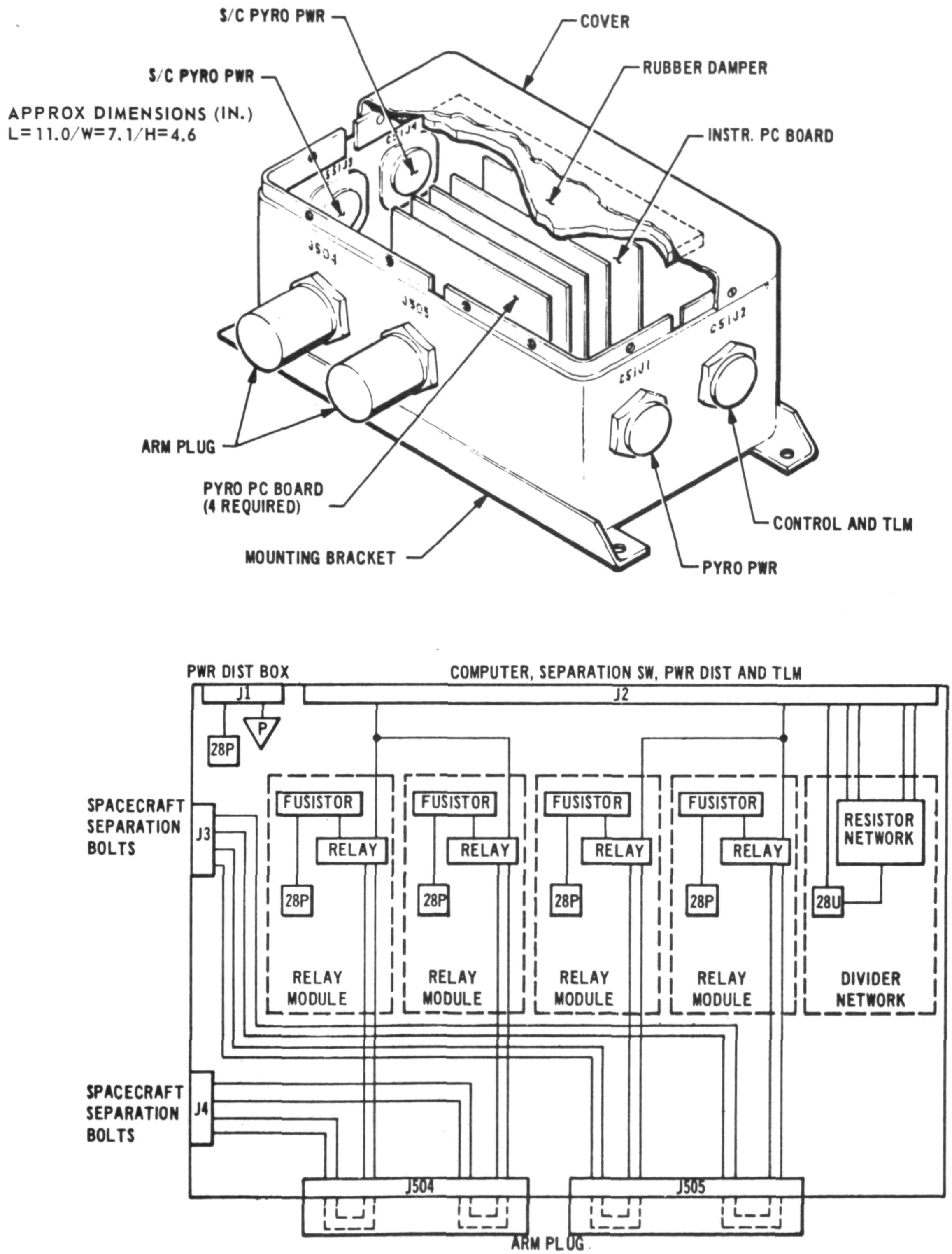


Fig. 2-33 Program Pyro and Monitor Box

2.3.2.4 Aft Control and Instrumentation Box. This unit (Fig. 2-34) is mounted in the Agena aft section and provides the electrical interface for controlling and monitoring the propulsion system components and other equipment located on the aft rack. It weighs approximately 6 lb.

The engine arm circuit (included in the engine control circuitry) consists of three latching relays, each operating from a separate command signal. The circuit is designed to operate upon receipt of any two of the three commands. This principle of operation is also used for engine shutdown. Blocking diodes in the oxidizer manifold pressure switch control circuits permit checkout of each of the two OMPS units during system test.

The propellant isolation valve circuits and the lip-seal/nitrogen regulator circuit each have four 5-amp nonlatching relays connected in a series-parallel combination for redundant reliability. These circuits are controlled by GC command and use unregulated power.

The signal-conditioning circuits accommodate the Agena aft section instrumentation requirements.

2.3.2.5 Discrete Control Box. This unit (Fig. 2-35) has the same external configuration and dimensions as the aft control and instrumentation box. It is located in the booster adapter and weighs approximately 7.5 lb.

The unit performs a variety of electrical interface functions involving the Agena, the booster, and the payload fairing. Redundant pyro circuits, using 5-amp nonlatching relays and protective fusistors, initiate payload fairing and booster separation events upon receipt of a GC command. Pitch and yaw commands, as well as discretes, are routed to the booster interface through fail-safe relay circuits. The pitch and yaw commands are signal-conditioned and monitored on telemetry. The discrete signals are routed directly to the telemetry unit.

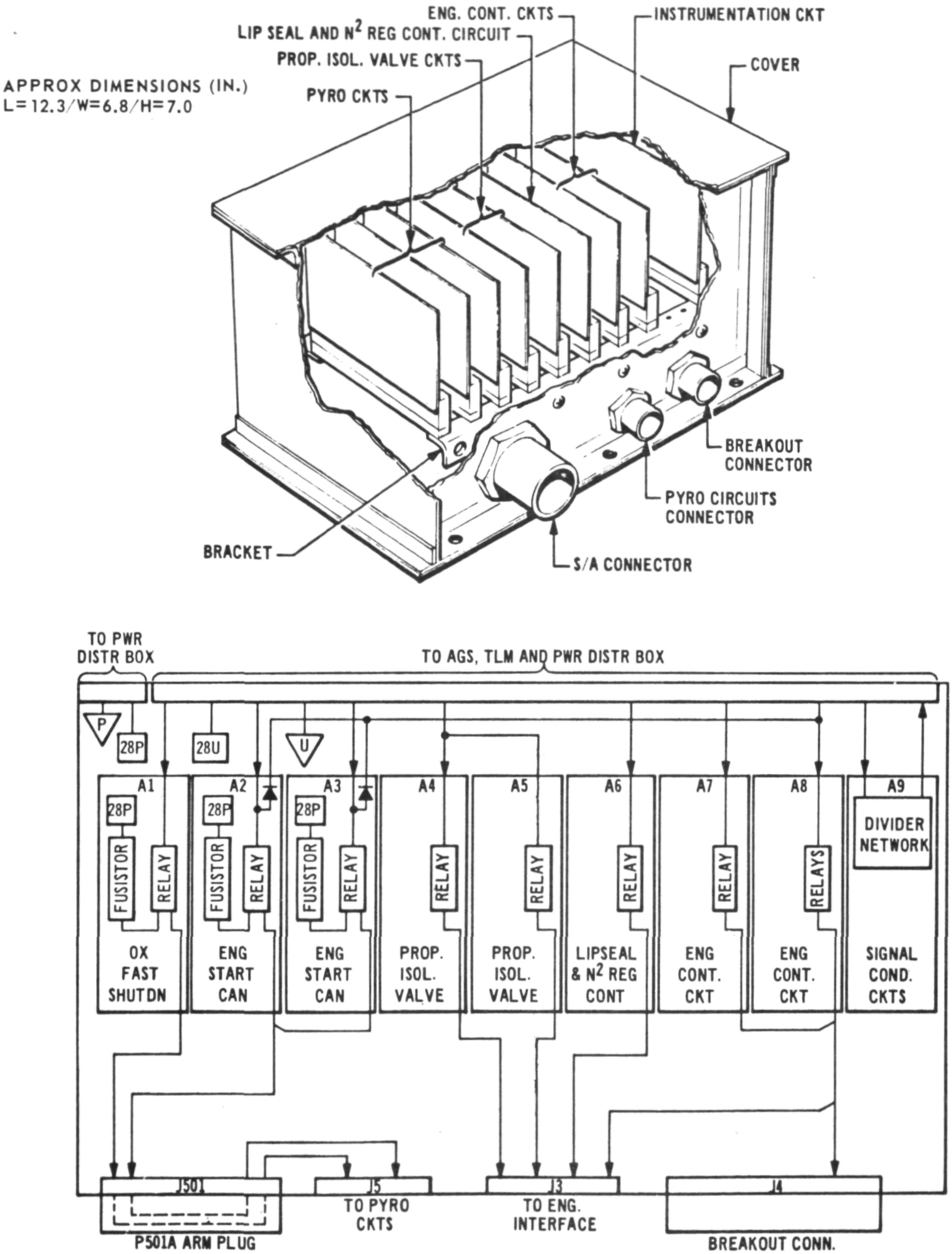


Fig. 2-34 Aft Control and Instrumentation Box

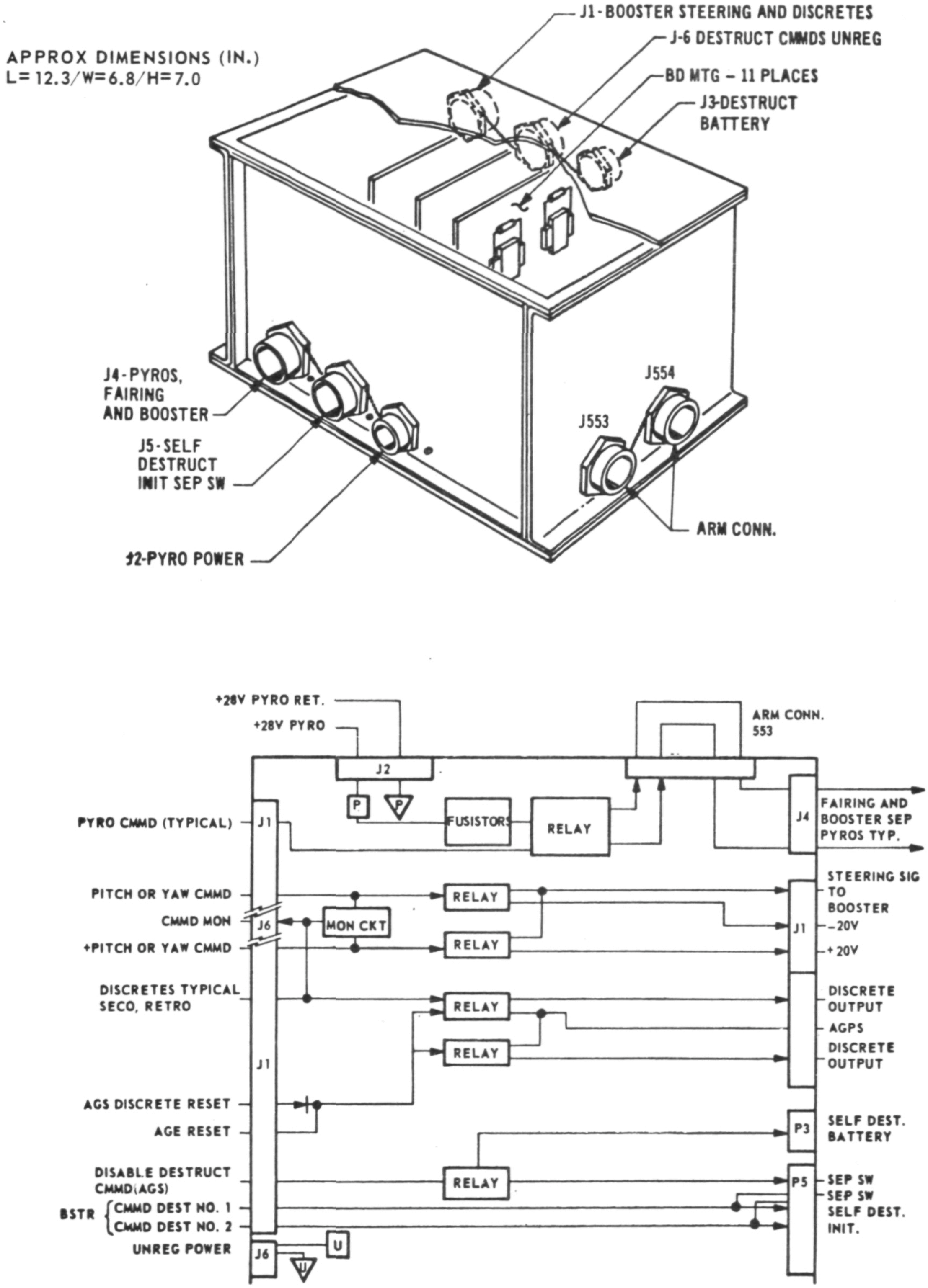


Fig. 2-35 Discrete Control Box

2.4 AVIONICS SYSTEM

The Agena ascent guidance and flight control and communications subsystems are discussed as the "avionics system" in the following pages.

2.4.1 Ascent Guidance System and Flight Controls

The Agena guidance and flight control system performs the vehicle guidance, control, and flight programming functions necessary to accomplish the mission. These functions include control of guidance steering, flight attitude, and the issuance of all discretes for vehicle system control, Agena engine ignition and cutoff, telemetry control, and payload separation.

The ascent guidance system (AGS) is designed to sense vehicle linear accelerations and angular rates and convert them to electrical signals. Using these signals and other electrical inputs, the AGS performs the following flight functions:

- Navigation. Compute Ascent Agena position and velocity with respect to a guidance reference frame at all times during flight.
- Guidance. Compute errors in flight with respect to a prescribed Ascent Agena reference trajectory, and issue compensating steering and thrust commands.
- Flight Control. Compute Agena body attitude and rate errors, compute compensation terms in the steering loop, and issue flight control commands to the Agena thrust vector controls and thrust valves.
- Programming. Issue programmed, time-sequenced commands.

AGS preflight functions include capability to program the AGS for checkout during systems test, vehicle testing at the launch pad, and flight-readiness verification at launch countdown.

The AGS (Fig. 2-36) consists of an inertial sensor assembly (ISA), a guidance computer (GC), flight control electronics (FCE), and pneumatic and hydraulic flight control systems. The AGS features a strapdown inertial reference system that senses vehicle motion by means of three rate-integrating gyros and three accelerometers that are part of the ISA. Outputs from the ISA are received by the guidance computer, which performs computations resulting in guidance commands. The vehicle coordinate system is shown in Fig. 2-37.

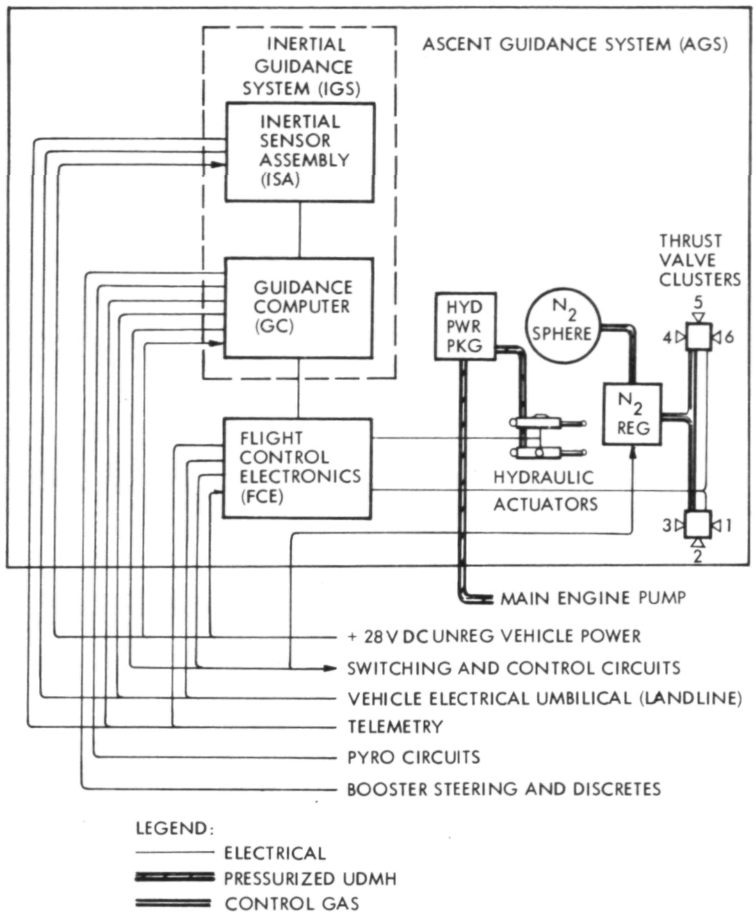


Fig. 2-36 Agena Ascent Guidance System

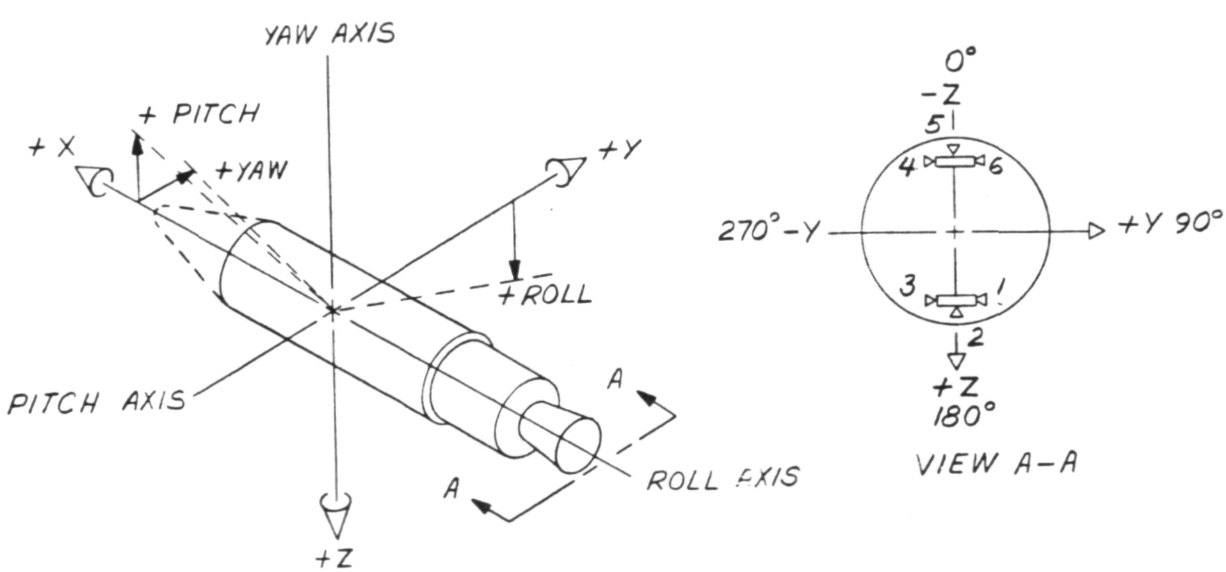


Fig. 2-37 Vehicle Coordinate System

2.4.1.1 Operating Characteristics. The AGS operates independently, without externally generated commands. During flight, accelerometers and gyros provide vehicle linear acceleration and angular rate measurements to the guidance computer. Prior to launch, the gyro reference orthogonal triad, which is fixed in vehicle coordinates, is determined in relationship with the computer reference coordinate system by means of prelaunch alignment procedures. Throughout flight, the gyro reference triad is rotated with the vehicle. As the vehicle turns, each single-degree-of-freedom gyro tends to precess from its null position. As soon as a preset output level is sensed by the threshold detector, a precise rebalance pulse is generated to drive the gyro back to its null position; hence, the gyro loops are pulse-on-demand loops. At the same time, a pulse of proper polarity is sent to the computer to indicate that the vehicle has rotated about the input axis of the gyro by the preset angular increment.

The three inertial accelerometers are mounted in an orthogonal triad that is colinear with the gyro reference triad. The pendulums of the accelerometers are torqued by pulses of alternate polarity back and forth through the null position, using a bang-bang type of rebalance scheme. When any accelerometer experiences an acceleration of the vehicle along the accelerometer's sensitive axis, the pendulum tends to be displaced from its null position; and the torque rebalance pulses switch to the polarity required to drive the pendulum to its null. With no acceleration applied along the accelerometer input axis, the total number of positive and negative torquing pulses over a period of time will be equal. However, if an acceleration is acting, there is a net difference in the number of rebalance pulses of one polarity. Each of the rebalance pulses is indicated to the computer as an incremental change in vehicle velocity.

2.4.1.2 Inertial Sensor Assembly (ISA). The ISA (Fig. 2-38) comprises an inertial sensor block, electronics, power supply, and connector housing, all physically mounted and electrically connected to a common baseplate.

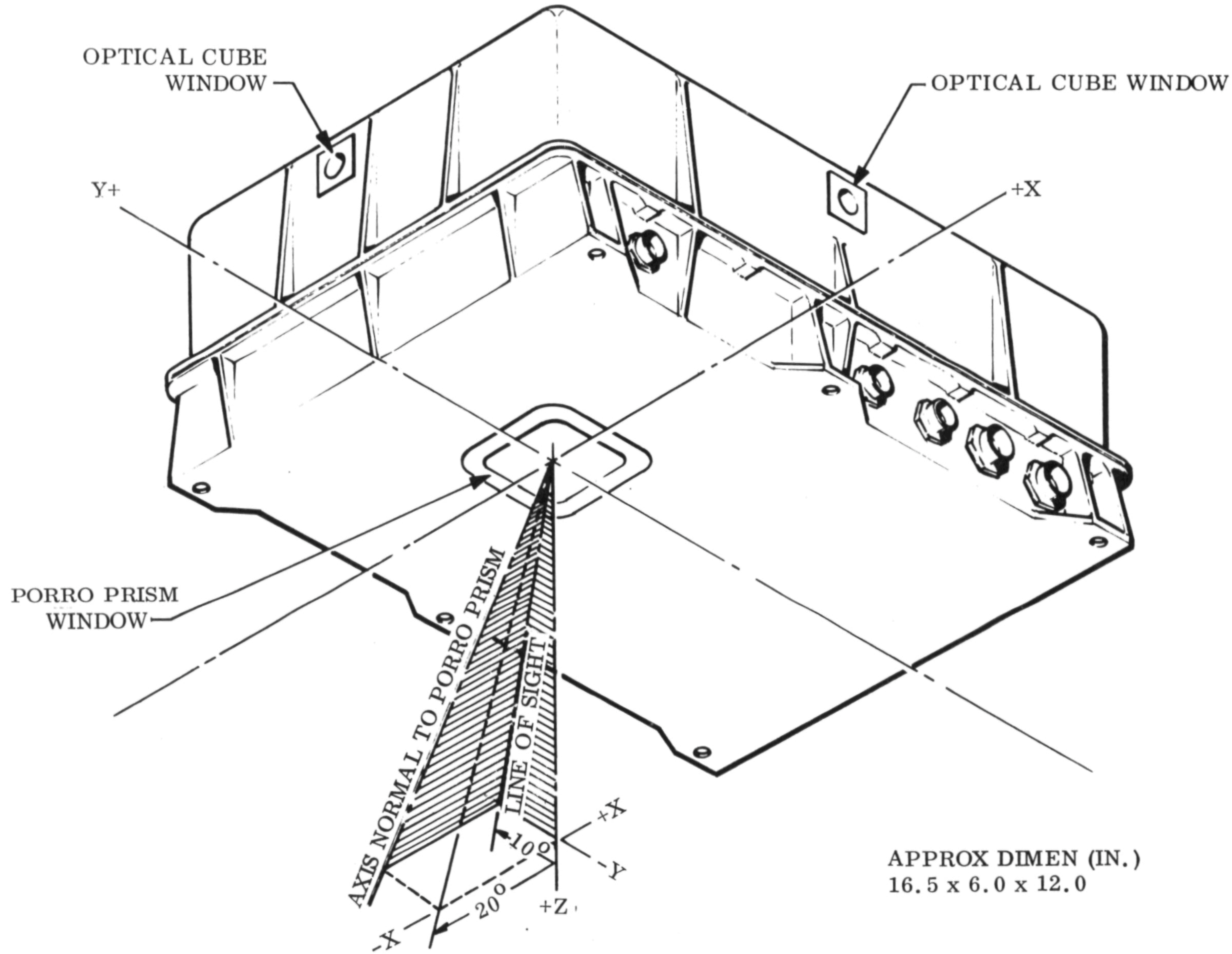


Fig. 2-38 Isometric View of ISA

It performs three-axis rate and acceleration measurements in a coordinate system related to the ISA mounting base. Rotational rates are determined up to 25 deg/sec with a quantization of 6.2947 arc-sec/pulse (low range) and 50.3576 arc-sec/pulse (high range). Linear accelerations up to 15 g are measured with the scale factor set at 0.27 ft/sec/pulse. The rebalance rate of the gyro and accelerometer loops is 1800 pulses/sec.

Inertial Sensor Block. The sensor block is a machined aluminum block containing three Honeywell GG334A8 integrating gyros, three Honeywell GG177P5 accelerometers, and the preamplifiers associated with each of the three gyro and accelerometer loops. Fast heaters, and thermostats are mounted on and within the block to maintain the block, and subsequently the accelerometers, at a constant operating temperature of 167°F. Gyro operating temperatures are maintained at +184°F by an individual temperature control amplifier and internal heater for each gyro. The prime frequency source, a 1.8432-MHz crystal, and the precision voltage reference are located in the block to take advantage of the constant temperature, thus assuring an extremely precise frequency standard for system timing. An optical cube mounted on the block, which can be viewed through three windows in the ISA housing, is the alignment reference for the system. A porro prism, also mounted on the block and viewed through a window in the base of the ISA, provides the reference surface for vehicle azimuth alignment on the launch pad.

Gyros. The angular rate sensing devices are Honeywell GG334A8 single-degree-of-freedom, integrating gyros. These gyros, an advanced design, are of the floated type and utilize a hydrodynamic gas bearing spin motor. Gimbal travel is restricted to less than 1 deg, thereby decreasing flexure of the leads between the case and gimbal. As a result, g-insensitive drift stability is improved. Since the gyro is operated near null and has an exceptionally low time constant (0.5 msec), induced drifts due to angular vibration environments are minimized. For the AGS application, gyro drift rates and stabilities are considered critical performance parameters and will meet the following requirements:

- G-insensitive drift ±0.3 deg/hr (maximum), 120-day stability ±0.25 deg/hr (3-sigma)
- G-sensitive drift ±2.0 deg/hr/g (maximum), 120-day stability ±0.6 deg/hr/g (3-sigma)

Accelerometers. Vehicle accelerations are measured by Honeywell GG177P5 accelerometers. These instruments, which are the flexure pivot type with fluid damping, have been used successfully on a number of Honeywell programs. The null bias is less than 200 x 10^{-6} g and is stable within 100 x 10^{-6} g for a 120-day period. As with gyro drifts, this is considered a key performance characteristic for the AGS application. Available test data show that these null bias figures are within the accelerometer capabilities.

Sensor Electronics Subassembly. The sensor electronics operate on the signals generated by the inertial components as a result of applied angular or linear motion. Additionally, the frequency division and gating from the prime frequency source are accomplished in the electronics subassembly and distributed to the entire ISA and to the guidance computer.

Signals from the inertial components are processed by the logic and gating circuits to control switching of the gyro rebalance current supply and accelerometer rebalance current supply outputs in the form of pulses of precise amplitude and width to rebalance the inertial components. Pulse outputs are furnished from the logic and gating function to the guidance computer.

The 1.8432-MHz prime frequency is counted down and, at intermediate steps in the division, frequencies are picked off and gated out for gyro and accelerometer loop control, power supply synchronization, and spin motor frequencies.

Four temperature control amplifier (TCA) circuits for controlling application of power to block heaters and gyro heaters complete the major functions contained as part of the sensor electronics. These circuits control unregulated 28 vdc to the heaters. If a block overtemperature condition occurs, a TCA will remove block heat by on-off control until the overtemperature condition ceases to exist. Three TCAs provide proportional heater control to the gyros.

Sensor Power Supply Subassembly. The power supply provides ac and dc power required by the sensor and electronics subassemblies. The input prime power, unregulated +28 vdc, is supplied through an input filter. Preregulation of the filtered voltage is accomplished in pulse-width modulated series switching regulators. The preregulated voltages are then regulated by dc converters.

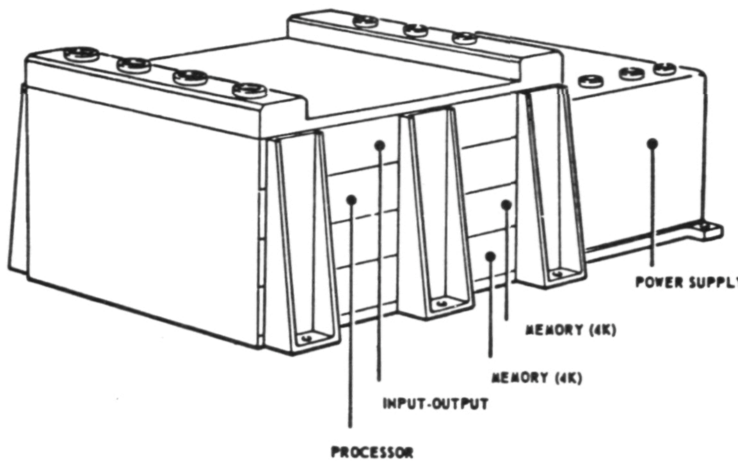
The gyro spin motor supply is a two-phase, 800-Hz square wave, 36 volts zero to peak to the three gyros. Signal generator excitation for the gyros and accelerometers is a 5-volt rms, 28.8-kHz sine wave. All ac frequencies are controlled by the prime frequency source.

The gyro spin motor power, as well as ISA main power, is controllable by external input signals from the computer. Gyro spin motor rotation detector circuits indicate when all gyro motors are running at 90 to 100 percent of synchronous speed.

2.4.1.3 Guidance Computer. The AGS guidance computer (Fig. 2-39) is an 8K, modified random-access coincident current core, 20-bit, parallel, fixed-point, two's complement, binary digital computer and consists of the following five sections:

1. Processor
2. Memory Bank A
3. Memory Bank B
4. Input/Output
5. Power Supply

The GC processes acceleration and angular rate data from the ISA, formats and outputs telemetry data, and issues steering signals and vehicle discretes as required for the mission. The GC has all of the characteristics of a general-purpose digital computer; it is the Honeywell SIGN III computer with program modifications. For the AGS application, the computer organization is represented as shown in Fig. 2-39. The arrangement in the illustration is hardware oriented and, in fact, reflects the actual modular construction of the GC.



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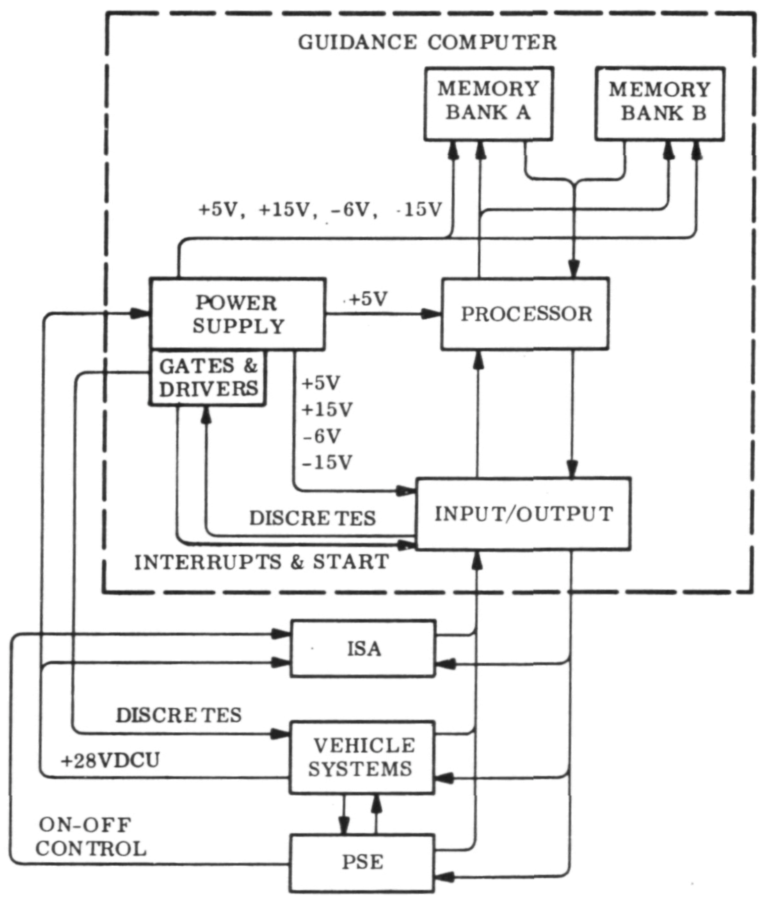


Fig. 2-39 AGS Guidance Computer

2.4.1.4 Flight Control Electronics. The AGS flight control electronics (FCE) is designed to utilize the specific capabilities of the IGS and to meet the specific needs of the mission. It is functionally divided into three categories: hydraulic channel, pneumatic channel, and buffer relays.

The FCE consists essentially of a spotwelded magnesium enclosure, three flat-cable connectors, four printed circuit component assemblies, and the four-layer matrix interconnect board. An isometric illustration of the unit with the cover removed is shown in Fig. 2-40. There are no wire harnesses or connectors within the FCE. The discrete and integrated circuit components are machine-soldered to the printed circuit boards, and the circuit board and flat cable headers are interconnected by split-wire wrap on the phosphor-bronze traces of the matrix board.

The FCE is approximately 6.5 in. high by 11.6 in. wide by 10 in. deep and weighs approximately 9 lb. Its unregulated 28-volt power consumption is about 3 watts.

The hydraulic channel consists of two identical servo amplifiers on a single card, one for pitch and one for yaw. The amplifiers are the differential-input type with two unipolar outputs each. Each output drives one of the two servomotor coils in the electrohydraulic servo actuator. Actuator position, indicated by the wiper of a wire-wound potentiometer on the actuator, is fed back to the summing point of the servo amplifier. This closes the inner loop, or servo loop, of the thrust vector control system. The amplifier is powered from the +20 volt regulator board in the FCE, which provides 0.25 percent regulation for amplifier operation and actuator excitation.

The pneumatic channel consists of six power gain stages (one to drive each thruster valve) with arc-suppression and associated components. The valve driver input has a threshold detector to eliminate false triggering.

The FCE contains three relays on each of the two valve driver boards. Four of these relays are used as isolation buffers between the GC and the program support equipment (PSE) during vehicle testing; i.e., the computer may command a relay and the PSE may monitor the contact closure, and vice versa. The other two relays are used in testing the pneumatic system.

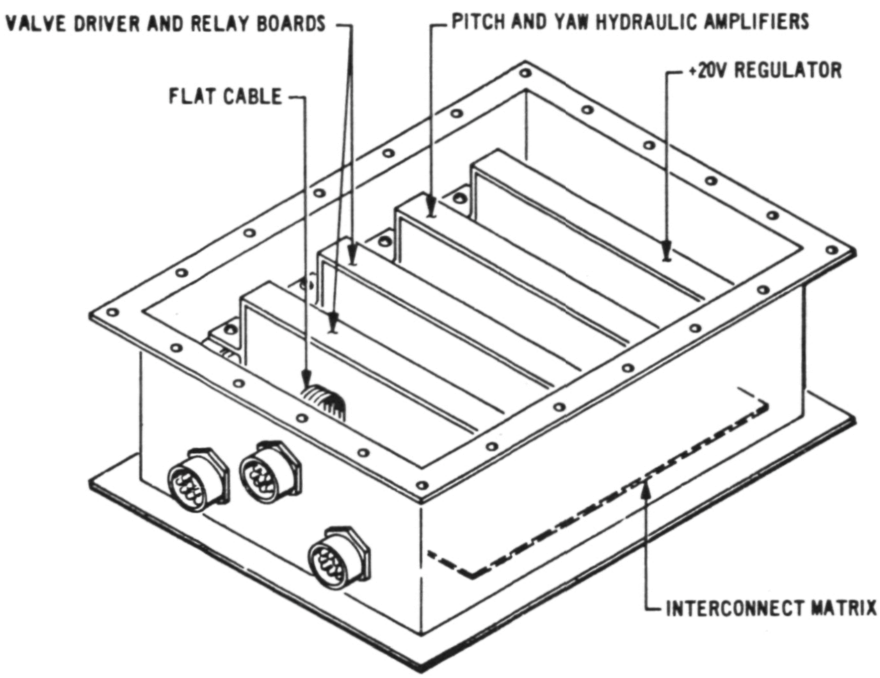


Fig. 2-40 Flight Control Electronics Assembly

2.4.1.5 Pneumatic Flight Control System. The pneumatic control system (Fig. 2-41) consists of a pneumatic pressure regulator, two thrust valve clusters, and a control gas storage sphere. The function of this system is to control the vehicle attitude by applying corrective forces to the Agena when attitude errors are sensed by the guidance system or programmed maneuvers dictate a change in attitude.

The pneumatic pressure regulator reduces the pressure of the gas fed to the thrust valves from the pressure in the supply sphere to a nominal 100 psia (high mode) or a nominal 5 psia (low-mode operations) and provides essentially constant regulation to the selected pressure while the thrust valves are pulsing and making demands on the gas supply. An additional function of the regulator is to supply low-pressure nitrogen gas to the Agena engine to pressurize the lip seal in the oxidizer turbine pump. This seal is pressurized to prevent oxidizer from seeping along the turbine shaft into the turbine gear case, where it could combine with fuel.

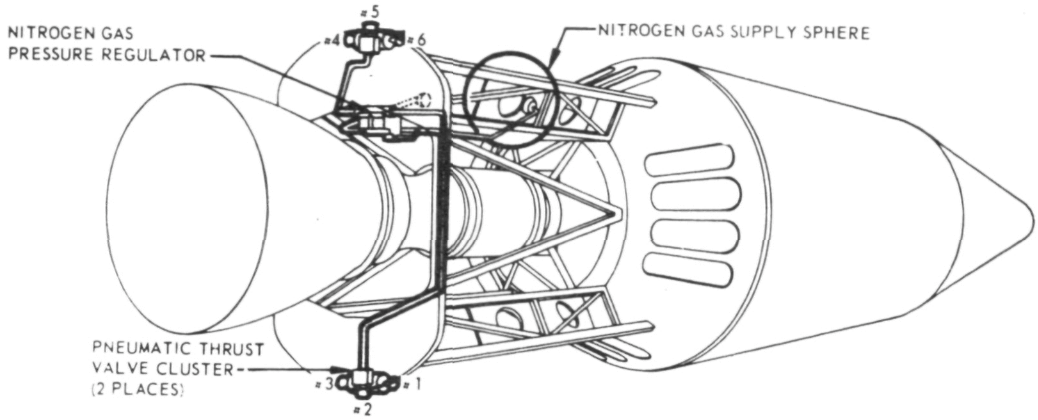


Fig. 2-41 Pneumatic Flight Control System

The solenoid-operated thrust valve cluster consists of three identical on-off solenoid valves mounted on a single manifold which contains a pressure port and electrical receptacle common to the three valves. The thrust controllers receive compressed gas from the nitrogen pressure regulator to provide 10 lb thrust at 100 psia in high mode and 0.5 lb thrust at 5 psia in low mode.

Each valve operates independently. Energizing the solenoid of a valve opens the valve, and the resulting gas flow produces thrust.

2.4.1.6 Hydraulic Flight Control System. The hydraulic control system (Fig. 2-42) consists of a hydraulic power package, servo actuators, and associated connecting parts. Thrust vector control is accomplished by gimballing the rocket engine thrust chamber by means of hydraulic actuators. Gimballing causes a displacement of the engine thrust direction from the Agena longitudinal axis, thereby forcing the Agena to change its flight attitude and direction. The actuators are controlled by signals from the flight control electronics. Hydraulic power to operate the actuators is supplied by a hydraulic power package that is driven by engine fuel (UDMH) pressure.

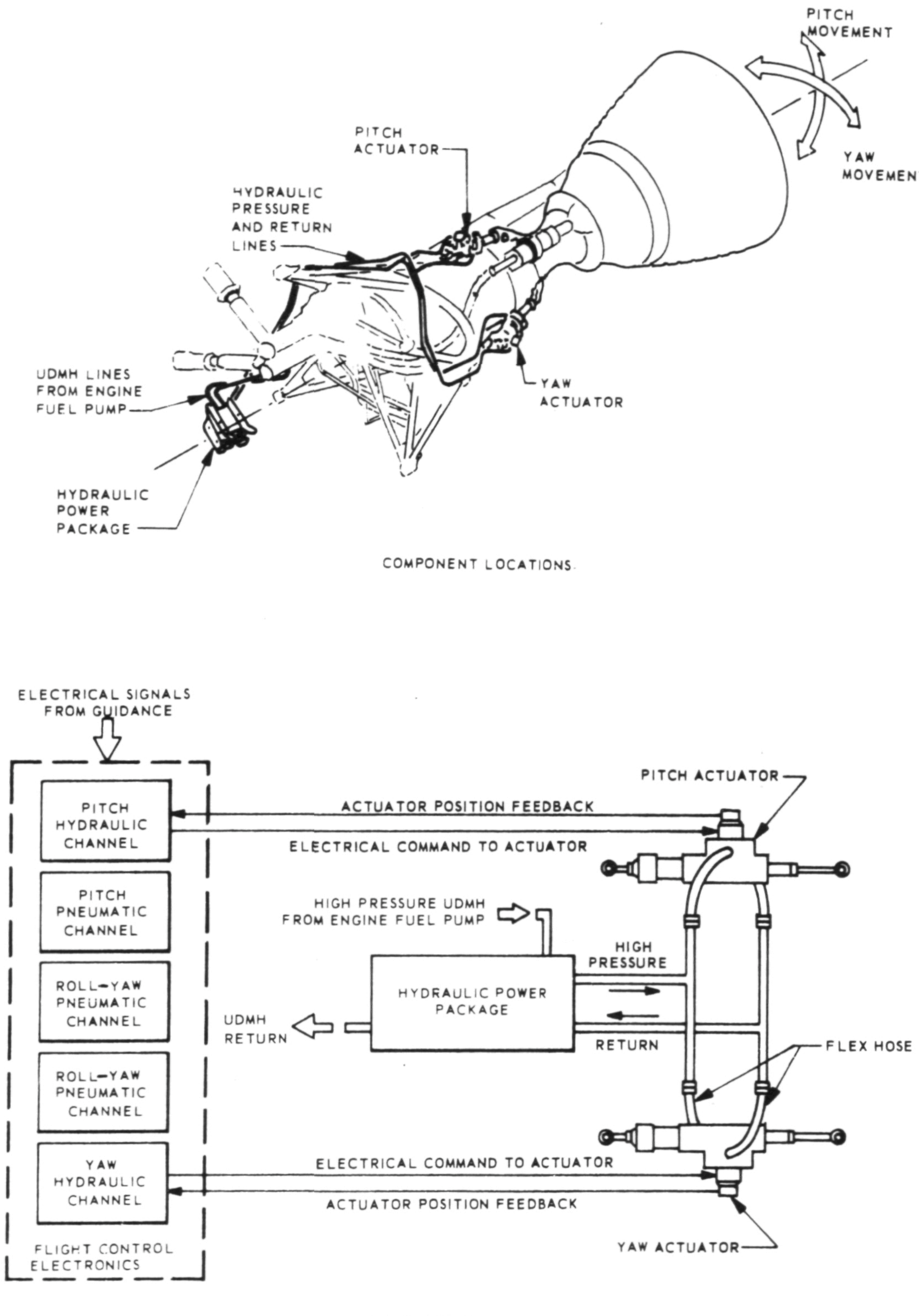


Fig. 2-42 Hydraulic Flight Control System

2.4.2 Communications System

This system provides the means by which vehicle data are monitored during system test and transmitted to ground stations during flight. As shown in Fig. 2-43, the system consists of a PCM telemeter for data encoding, a tape recorder for acquisition of stored data, a baseband assembly unit to provide two data channels, and an S-band transmitter and antenna system.

No equipment is provided specifically for tracking, which is accomplished through acquisition of the telemetry carrier frequency by ground station antennas. Skin tracking by radar can be used during ascent. Also, there are no realtime commands. Stored program commands for executing vehicle functions are supplied by the AGS guidance computer.

The Agena uses two telemetry channels for realtime data acquisition by ground stations. One channel is used by the PCM telemeter system in collecting and processing various Agena data, including pressure, acceleration, and temperature measurements; AGS computer computations; and other essential vehicle information. The other channel is used in monitoring high-density AGS pulse data for evaluation of AGS performance and in transmitting stored data from the tape recorder. The AGS data are transmitted continuously over this channel until the tape recorder output is switched to it by the telemetry junction box.

The PCM telemeter accepts analog, bilevel, and digital inputs, converts the analog signals to digital form, and combines them with the bilevel and digital inputs into a non-return-to-zero-level (NRZ-L) digital bit stream output. The analog signals are from the various status transducers in the vehicle. The bilevels are from various event monitors, such as relay switch closures. A counter converts the turbine speed

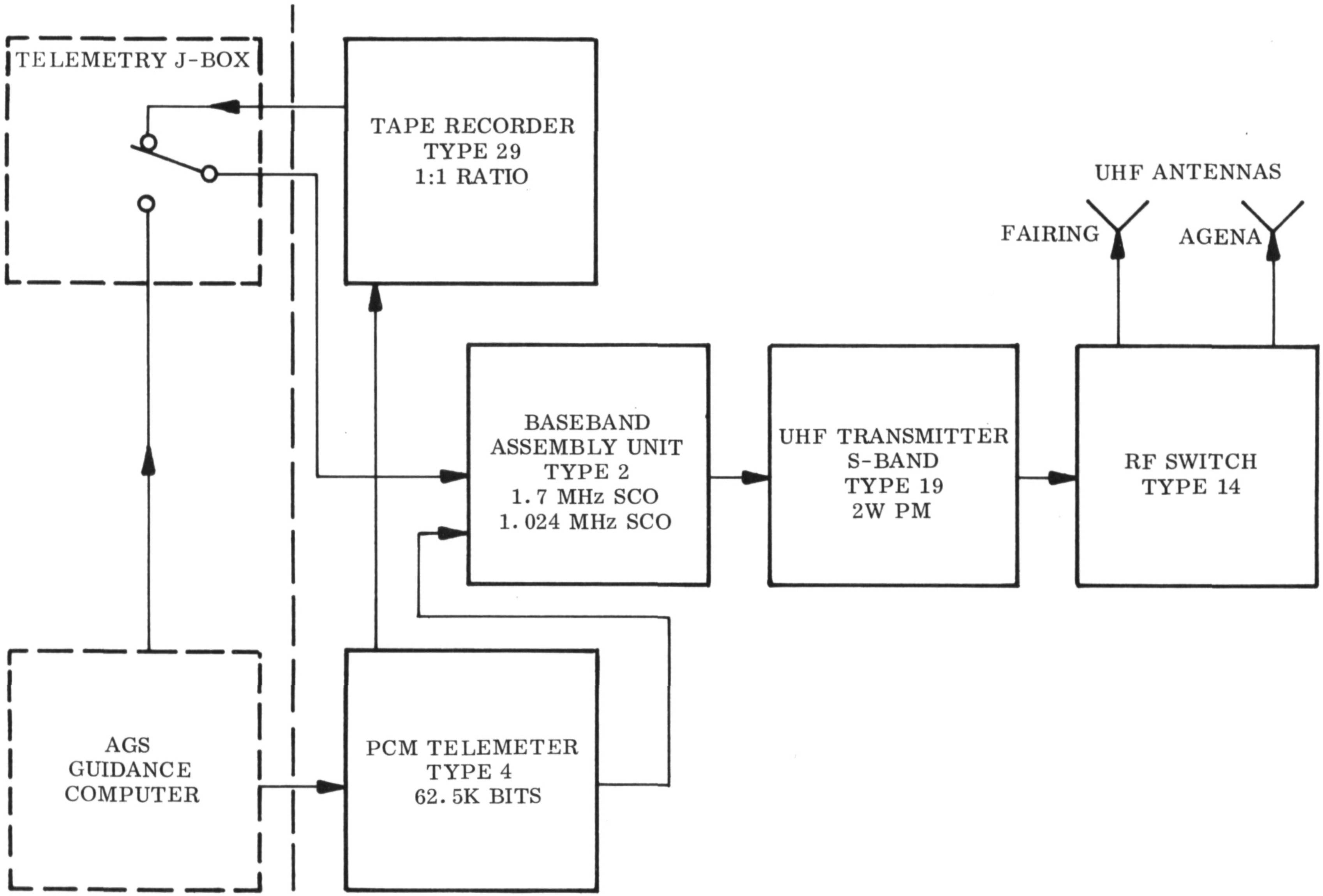


Fig. 2-43 Communications System Block Diagram

data into digital form. Computer readout in digital form comes from the AGS system and are inserted into the bit stream at the proper time as controlled by the telemeter clock, which is synchronously driven by the AGS GC clock.

Two outputs from the telemeter are provided (see Fig. 2-43). One is filtered and supplied to the input of the baseband assembly unit. The other is nonfiltered and is recorded by the tape recorder. Of the two telemetry channels in the baseband assembly unit, one channel is established through a 1.024-MHz subcarrier oscillator that is biphase-modulated by the PCM telemeter; the other channel is established through a 1.7-MHz SCO that is biphase-modulated first by high-density AGS pulse data and then by data from the tape recorder. The outputs of these SCOs are combined in a summing amplifier with a manual level control to adjust the deviation of the phase-modulated transmitter. These SCO frequencies are compatible with the SGLS ground stations used with the program.

The RF system consists of two S-band antennas, an RF switch, and the phase-modulated UHF transmitter. Two antennas are required because the Agena-mounted unit is covered by the payload fairing during ascent. The ascent antenna is mounted on the booster adapter and is connected to the transmitter output by an RF switch. The RF switch is in the ascent antenna mode when power is applied to it, and in the Agena antenna mode when power is removed. Power is removed from the RF switch through the disconnection of connector P700 at booster separation. The Agena antenna is then connected to the RF system. Output of the transmitter is 2 watts minimum at 2200 to 2300 MHz. The transmitter is phase-modulated to be compatible with the SGLS ground stations.

The tape recorder is used to store telemetry data during critical periods when no real-time acquisition stations are available. The AGS GC commands the tape recorder to record data for 2 min at the end of Agena first burn and 8 min around second burn, for a total of 10 min recording time. Playback occurs by GC command when the vehicle is over a tracking station.

2.4.2.1 Instrumentation Data. The vehicle status information, AGS inputs, and other instrumentation data monitored by the telemetry system are transmitted in real time to ground stations. One group of AGS data (designated AGS-1) originates at the GC; these data are essentially the computer computations and are digital word inputs to the PCM telemetry. Another group of AGS data (designated AGS-2) originates at the inertial sensor assembly; these data are the high-density AGS raw gyro and accelerometer pulse data. The functions of formatting and multiplexing the AGS-1 data are performed by the Type 4 PCM telemeter. The AGS-2 data formatting and multiplexing functions are performed by the ISA. Additional AGS monitors are provided for temperatures, voltages, and spin motor rotation detection (SMRD).

2.4.2.2 PCM Telemeter. The Type 4 PCM telemeter consists of a timing and control system, analog multiplexers, bilevel multiplexer and decision amplifier, direct digital circuitry, analog-to-digital converter, output filter, and power supply.

The Type 4 unit is of modular construction, with each module being soldered to its position on one of the four printed circuit boards. Interboard wiring and board-to-connector wiring is done by flexible cables. The unit weighs 2 lb, and its dimensions are approximately 4 by 4 by 3 in.

2.4.2.3 Tape Recorder. The Type 29 tape recorder is a two-reel recorder/reproducer used to store operational and status data in a PCM format for later transmission to ground stations. It is a single-track recorder capable of recording in only one tape direction and reproducing in either tape direction. Recording capability is 10 (+2, -0) min. Reproduce time is also 10 min; i.e., the record/reproduce speed ratio is 1:1. The bit rate of the recorder is 62,500 bits/sec, in NRZ-L digital format.

Operation of the Type 29 is controlled by three externally generated commands: record, reproduce, and off. In the record mode, the unit records until one of two events occurs: an off command is received or the unit reaches its end-of-tape sensor. The unit stops in either case; however, if the off command initiated the stop event, the unit will respond

to either a subsequent record or reproduce command. If the unit has reached its end-of-tape sensor, it responds to a reproduce command only. In the reproduce mode, the unit automatically recycles each time it reaches its end or beginning of tape sensor. The unit continues to reproduce until an off command has been given.

The Type 29 recorder weighs approximately 12 lb; its dimensions are 10.5 by 8.0 by 3.8 in. The unit consumes an average of 15 watts power, with a peak power of 26 watts each time a new command is given or the automatic recycle period is reached.

2.4.2.4 Baseband Assembly Unit. The Type 2 baseband assembly unit accepts two PCM input signals, each of which biphase-modulates a subcarrier. The subcarriers are combined into a composite frequency-multiplexed output.

Nominal dimensions of the unit are 1.76 by 3.63 by 5.21 in.; nominal weight is 2 lb. The unit has three major functional sections: the oscillator/modulator, summing amplifier/output driver, and regulator and power control.

2.4.2.5 UHF Transmitter. The Type 19 UHF transmitter is a phase modulation transmitter operating in the S-band region. It weighs 4 lb (maximum), and its nominal dimensions are 2.0 by 3.83 by 7.44 in. It is a solid-state device capable of operating in the ground checkout, ascent, and space environments of the Agena vehicle. The unit is required to develop 2.0 or more watts, throughout the supply voltage and environmental ranges, while operating into a nominal 50-ohm load having a maximum voltage standing wave ratio (VSWR) of 2.0 to 1 at any phase. An open or short circuit at the RF output will not cause degraded performance once these conditions are removed.

The transmitter input impedance is 10,000 ohms minimum, shorted by 50 pf maximum. The unit will accept a signal or signals within the frequency range of 100 Hz to 2 MHz at levels up to 6 volts peak-to-peak. These signals linearly phase-modulate the RF carrier. Overvoltages on the input at levels up to 10 volts peak-to-peak will not damage the unit.

2.4.2.6 S-Band Antenna. The S-band antenna is a one-quarter wavelength, monopole, omnidirectional, linear polarized antenna designed to operate within the frequency range of 1750 to 2300 MHz. Nominal dimensions are 2.5 in. in diameter by 2.08 in. high, and the maximum weight is 12 oz.

The relatively simple antenna construction consists of four major parts:

1. RF coax connector
2. Aluminum radiator element (0.5 in. in diameter by 1.5 in. long)
3. Teflon cover (1.5 in. in diameter by 2.1 in. high)
4. Steel baseplate (2.75 in. in diameter)

The antenna is installed on the vehicle by attaching the baseplate to the vehicle skin. The steel base is threaded so that a hat coupler may be installed on the antenna during ground checkout operations.

2.4.2.7 RF Switch. The Type 14 RF switch is a single-pole, double-throw, fail-safe coaxial device used for transferring RF energy between two loads. The nominal dimensions are 1.1 by 2.75 by 3.25 in., and the weight is less than 0.6 lb.

Section 3

VEHICLE INTERFACES

Section 3
VEHICLE INTERFACES

3.1 PAYLOAD ADAPTATION

Payload adapters to mount the payload to the Agena are normally designed for the distinct requirements of each using program. However, it may be feasible for a potential user of an Ascent Agena to adapt to an existing or modified existing payload/ Agena adapter.

Adapter structural design, which must meet attachment load criteria, also generally embodies provisions for payload separation impulse (force). Separation is commonly achieved with spring mechanisms. Payload-to-adapter restraint is accomplished with pyrotechnically actuated explosive bolts, nuts, pinpullers, or explosively severed joints.

On the Agena adapter assembly shown in Fig. 3-1, four payload separation spring (ejection) assemblies and a pyro-released V-band payload restraint are used. An alternate approach to payload adaptation is shown in Fig. 3-2. The existing payload-to-Agena mounting interface provisions are illustrated in Fig. 3-3.

3.2 GROUND DISCONNECTS AND ACCESS DOORS

The Ascent Agena umbilical connection and access panel locations for prelaunch operations are shown in Fig. 3-4. The payload fairing and booster adapter are also shown for reference.

3.3 AIRBORNE DISCONNECTS

In some recent typical configurations, the Ascent Agena has three connectors across the booster adapter interface. These provide steering commands to the booster from

the ascent guidance system, Agena ascent telemetry RF to an ascent antenna located on the booster adapter, and discrete commands to the separation and self-destruct systems, also in the booster adapter. These three connectors disconnect when the Agena separates from the spent booster. In addition, three connectors are provided across the Agena/spacecraft (payload) interface from the program pyro and monitor box; these disconnect upon separation of the payload from Agena.

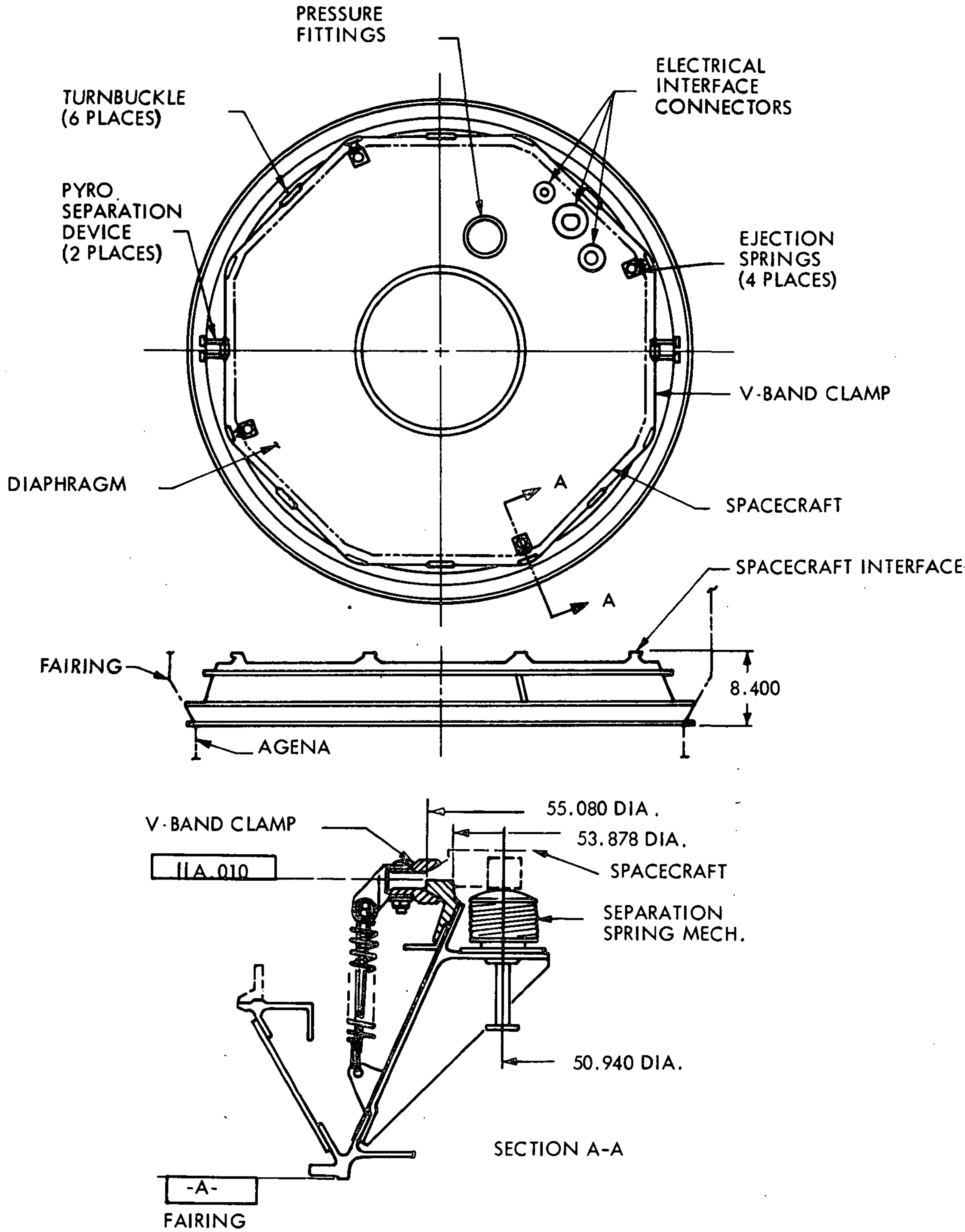


Fig. 3-1 Payload/Agena Adapter

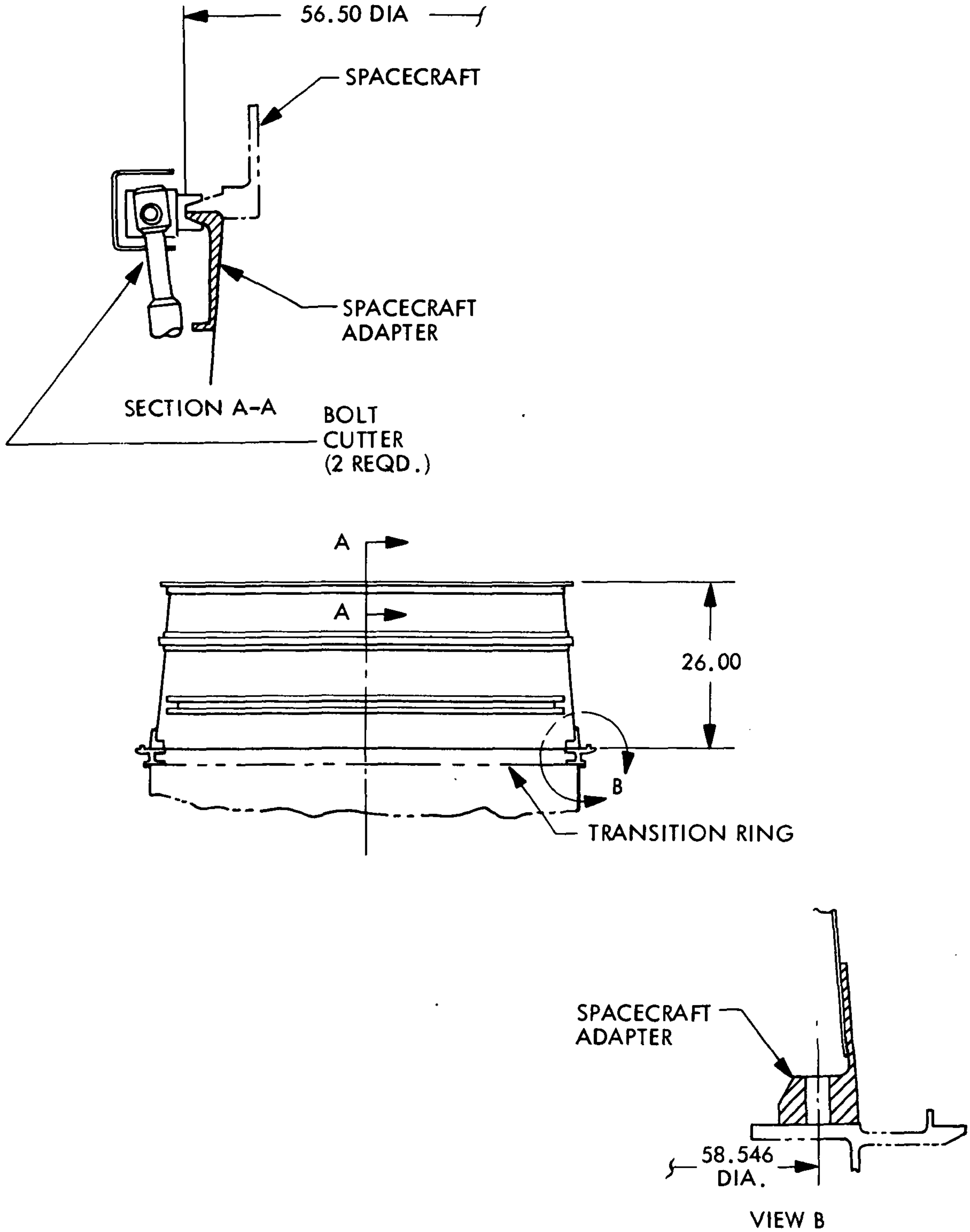
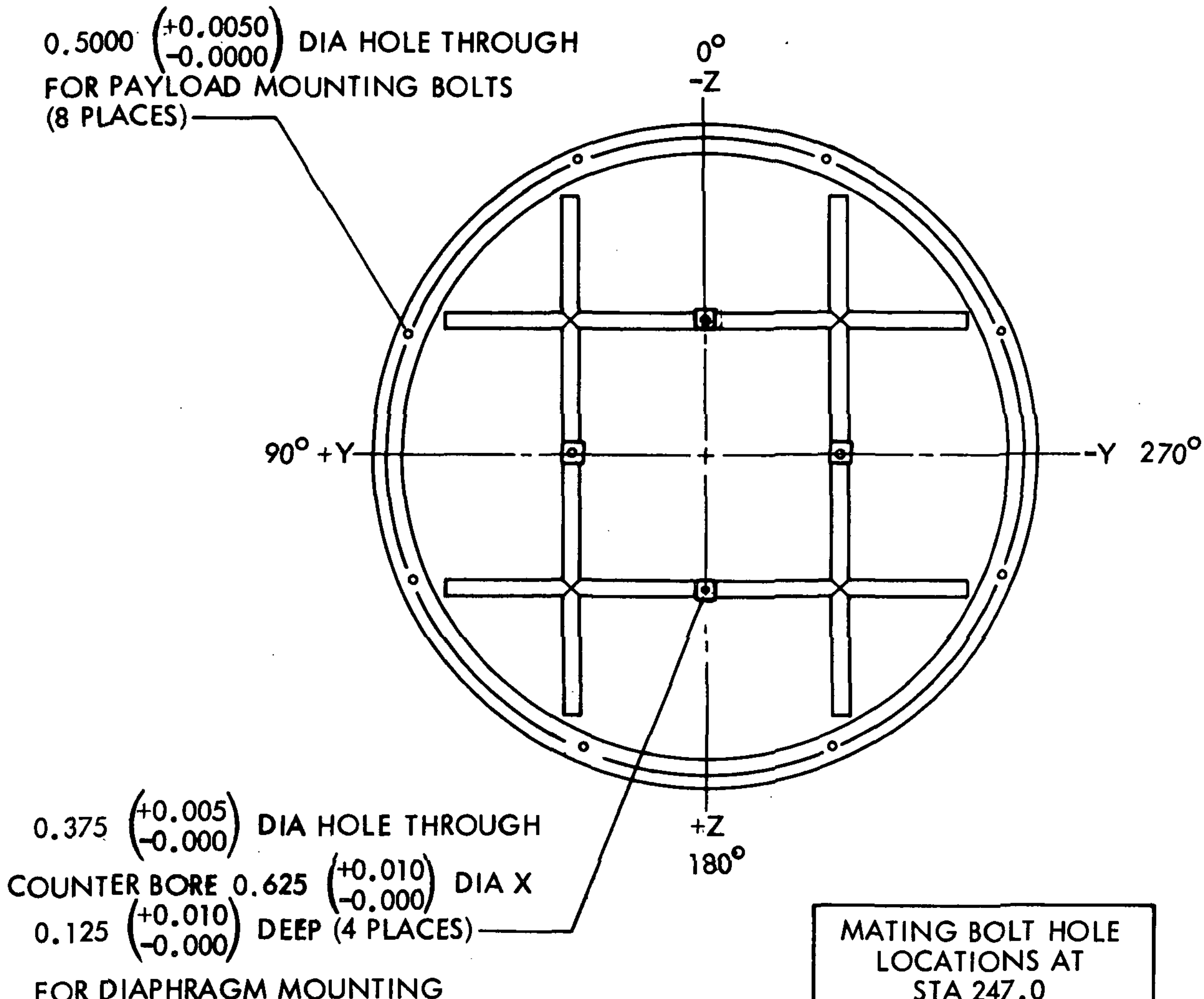


Fig. 3-2 Alternate Payload/Agena Adapter



NOTE: 1. RECOMMENDED BOLT FOR FULL STRENGTH:
DIAMETER 0.4995 $\binom{+0.0000}{-0.0010}$ AFTER PLATING
2. HEAT TREAT 180,000 - 200,000 PSI ULTIMATE TENSILE STRENGTH
3. TORQUE 480-690 IN.-LB

MATING BOLT HOLE LOCATIONS AT STA 247.0	
DEGREES	RADIUS (IN.)
22°29'59"	29.006
67°30' 9"	29.009
112°29'55"	29.010
157°29'49"	29.010
202°29'33"	29.010
247°29'24"	29.009
292°29'33"	29.006
337°30' 3"	29.007

Fig. 3-3 Payload/Agena Mounting Interface Hole Locations

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EQUIPMENT ACCESS/LOCATION				
REF.	EQUIPMENT OR FUNCTION	LOCATION	ACCESS	REMARKS
10	SAFE PLUG/ARM PLUGS AND TEST PLUGS ACCESS	BAY 1	REMOVE FORWARD DOOR	
	FORWARD SAFE/ARM BOX	BAY 1	REMOVE AFT DOOR	
	HELIUM PRESSURIZATION SYSTEM	BAY 1	REMOVE AFT DOOR	
	BATTERIES, POWER DISTRIBUTION "J" BOX, POWER MODULE, AMP HOUR METER	BAY 2	REMOVE BAY 2 DOOR	
11	SENSOR HEAD (RIGHT HAND) SENSOR HEAD (LEFT HAND)	BAY 3 BAY 5	REMOVE RIGHT SENSOR PANEL REMOVE LEFT SENSOR PANEL	
	GUIDANCE MODULE	BAY 3,4 & 5	REMOVE RIGHT SENSOR PANEL/BAY 3 REMOVE LEFT SENSOR PANEL/BAY 5	SEE GUIDANCE MOD. PROFILE FIG.
	ADJUST ALPHA ANGLE	BAY 4	REMOVE HAND HOLE COVER IN GUIDANCE MODULE SKIN	
	S-BAND BEACON ANTENNA	BAY 4	ACCESS FROM INBOARD SIDE OF GUIDANCE MODULE	
	C-BAND BEACON ANTENNA	BAY 4	ACCESS FROM INBOARD SIDE OF GUIDANCE MODULE	
	FLIGHT CONTROL ELECTRONICS, FLIGHT CONTROL "J" BOX & SEQUENCE TIMERS	BAY 6	REMOVE BAY 6 DOOR	
	ORBITAL PROGRAMMER COMMAND DESTRUCT MODULE	BAY 7	REMOVE BAY 7 DOOR	
	BEACON AND TELEMETRY MODULE	BAY 8	REMOVE BAY 8 DOOR	
	COMMAND DESTRUCT ANTENNA	BAY 1 BAY 7	ACCESS THRU BAY 1 ACCESS THRU BAY 6	
12	VHF ANTENNA (TYPE XIII)	BAY 4	REMOVE LOWER TANK FAIRING & AFT SKIN BAY 4	REMOVE LOWER TANK FAIRING & AFT SKIN BAY 4 FOR S-01B-62 & UP
13	COMMAND DESTRUCT UNIT	STA 313.35 @ 1°54'	REMOVE COMMAND DESTRUCT DOOR	
14	AFT SAFE/ARM BOX AFT & SEPARATION SAFE/ARM PLUG	STA 402.00 @ 195°0'	OPEN SAFE/ARM DOOR	
15	SELF DESTRUCT CHARGE/ BATTERIES	STA 403.25 @ 294°0'	REMOVE SELF DESTRUCT CHARGE FAIRING	
16	PROPELLANT FILL COUPLING (OXIDIZER)	STA 409.02 @ 87°0'	OPEN PROPELLANT FILL DOOR	
17	PROPELLANT FILL COUPLING (FUEL)	STA 409.02 @ 110°30'	OPEN PROPELLANT FILL DOOR	
5	GAS FILL VALVE (NITROGEN)	STA 391.06 @ 98°30'	OPEN NITROGEN FILL DOOR	
18	FORWARD ROLLER ADJUSTMENT	STA 415.00 @30°0', 150°0', 210°0' & 330°0'	OPEN ROLLER ADJUST DOORS	
19	AFT ROLLER ADJUSTMENT	STA 463.00 @ 30°0', 150°0', 210°0' & 330°0'	OPEN ROLLER ADJUST DOORS	
20	RETRO ROCKET	STA 468.90 @ 0°0' & 180°0'	REMOVE RETRO ROCKET FAIRINGS	

ACCESS DOOR LOCATIONS			
REF.	DESCRIPTION	VEHICLE STATION	VEHICLE LOCATION ANGLE
21	BTL UMBILICAL CUT OUT COVER	280.80	84°
22	DORSAL ANTENNA COVER	280.50	15° & 324°
23	STRETCH SLING FITTING	448.61	90° & 270°
24	ELECTRICAL INTERFACE PLUG AND ACCESS DOOR	501.71	285°
25	RATE GYRO ACCESS DOOR	503.71	255°
26	LOX VENT	503.85	136°30'

DISCONNECT LOCATIONS (ELECTRICAL CONNECTORS AND MECHANICAL COUPLINGS)			
REF.	DESCRIPTION	VEHICLE STATION	VEHICLE LOCATION ANGLE
1	MAIN ELECTRICAL UMBILICAL (J100)	251.75	101°
10	ELECTRONICS TEST CONNECTOR (J200)	250.94	41°
10	GUIDANCE & CONTROL TEST CONNECTOR (J201) TEST CONNECTOR (J201)	252.54	47°
	INERTIAL REFERENCE PACKAGE HEATER (J300)	253.25	192°30'
10	FORWARD SAFE/ARM CONNECTOR (J500)	250.44	33°30'
	AFT SAFE/ARM CONNECTOR (J502) AFT SAFE/ARM CONNECTOR (J501)	400.76	256°
	2ND BURN SAFE/ARM CONNECTOR (J503)	400.76	263°
	SEPARATION SAFE/ARM CONNECTOR (J503)		
2	PROPELLANT VENT COUPLING (OXIDIZER)	251.75	72°30'
3	PROPELLANT VENT COUPLING (FUEL)	251.75	88°30'
4	GAS FILL VALVE (HELIUM)	276.50	107°
5	GAS FILL VALVE (NITROGEN)	391.06	98°10'
6	AIR CONDITION COUPLING ASSEMBLY	280.00	124°
16	PROPELLANT FILL COUPLING (OXIDIZER)	404.34 409.02	87°
17	PROPELLANT FILL COUPLING (FUEL)	404.34 409.02	110°30'
7	OXIDIZER DRAIN	428.14	146° 153°
8	FUEL DRAIN	422.14	143° 157°
9	SNIFFER HOLE (FUMES DETECTOR)	489.21	102°

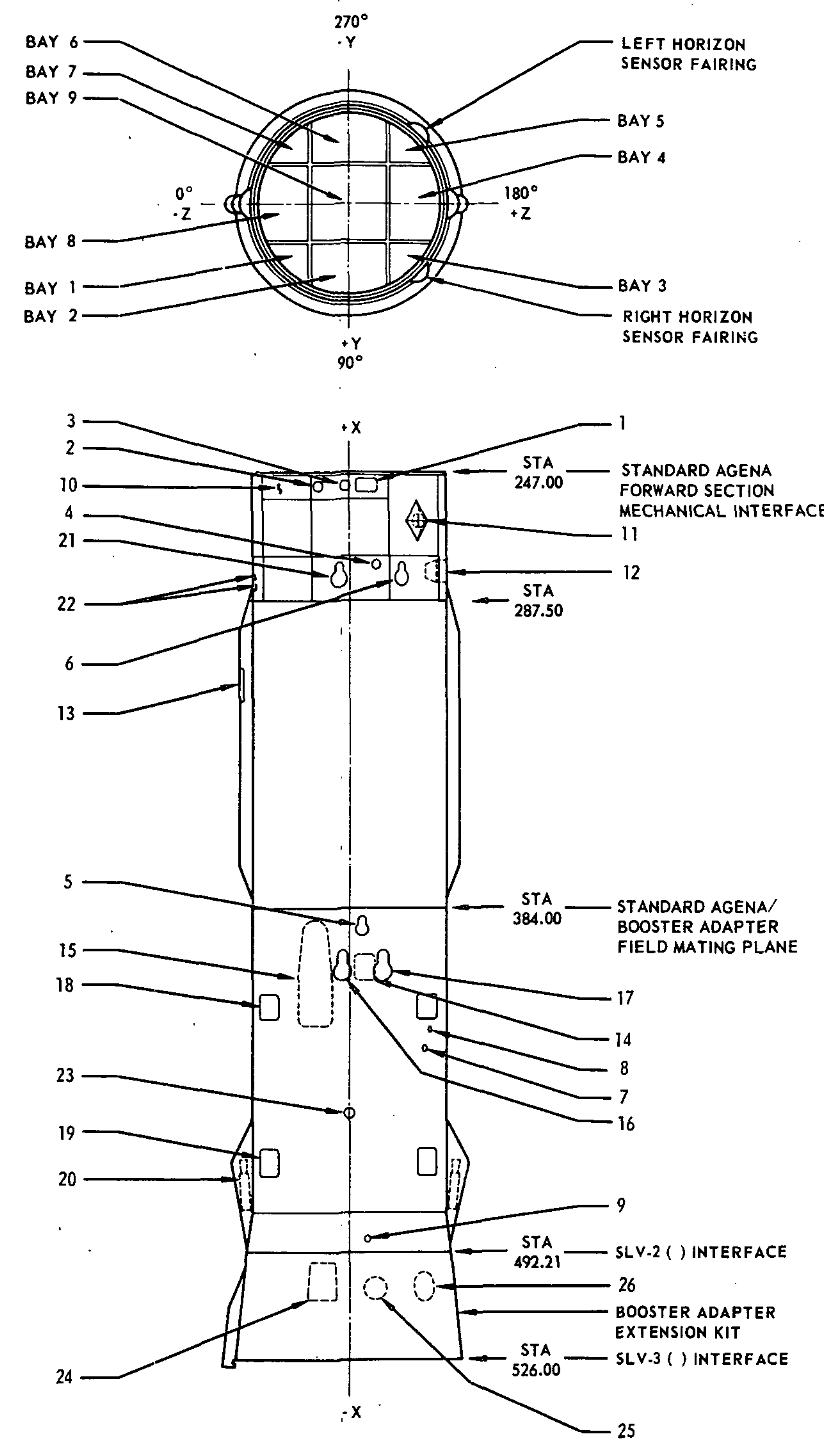


Fig. 3-4 Typical Spaceframe Interconnect Provisions

Section 4

ENVIRONMENTAL CONSTRAINTS AND TEST REQUIREMENTS

Section 4
ENVIRONMENTAL CONSTRAINTS AND TEST REQUIREMENTS

Environmental conditions to be considered when application of the Agena is under study are outlined in this section. Since specific mission applications involve many variables, the information presented is of necessity, general in nature. Data on specific booster/Agena systems are beyond the scope of this document. Such data, however, have been thoroughly documented in other currently available reports.

Analysis of the most probable as well as the most severe flight environments can provide realistic design and test input levels for the Agena/payload configuration. Flight environments for Agena flights on different boosters may be employed in assessing the excitation levels that may be experienced. For shock, acoustic, and transient environments, levels have been established by enveloping applicable flight data. For other environments, such as random vibration and pure sinusoidal POGO oscillations (most prominent on Thorad), the data have been derived statistically.

The normal flow and interrelation of supporting analysis tasks routinely accomplished by LMSC for an integrated vehicle/spacecraft design are shown in Fig. 4-1. Effort is normally initiated in a definition phase during which mission requirements, structural criteria, environments and environmental requirements, and preliminary mass and aerodynamic data are established. Preliminary reference and design trajectories are generated consistent with requirements and are used to provide criteria to the spacecraft designer for preliminary design. After a preliminary launch vehicle/spacecraft design is established, analyses can then be performed to determine design margins and to identify test requirements. Finally, following a prelaunch wind analysis, control and bending parameters are reviewed to determine ascent vehicle launch readiness.

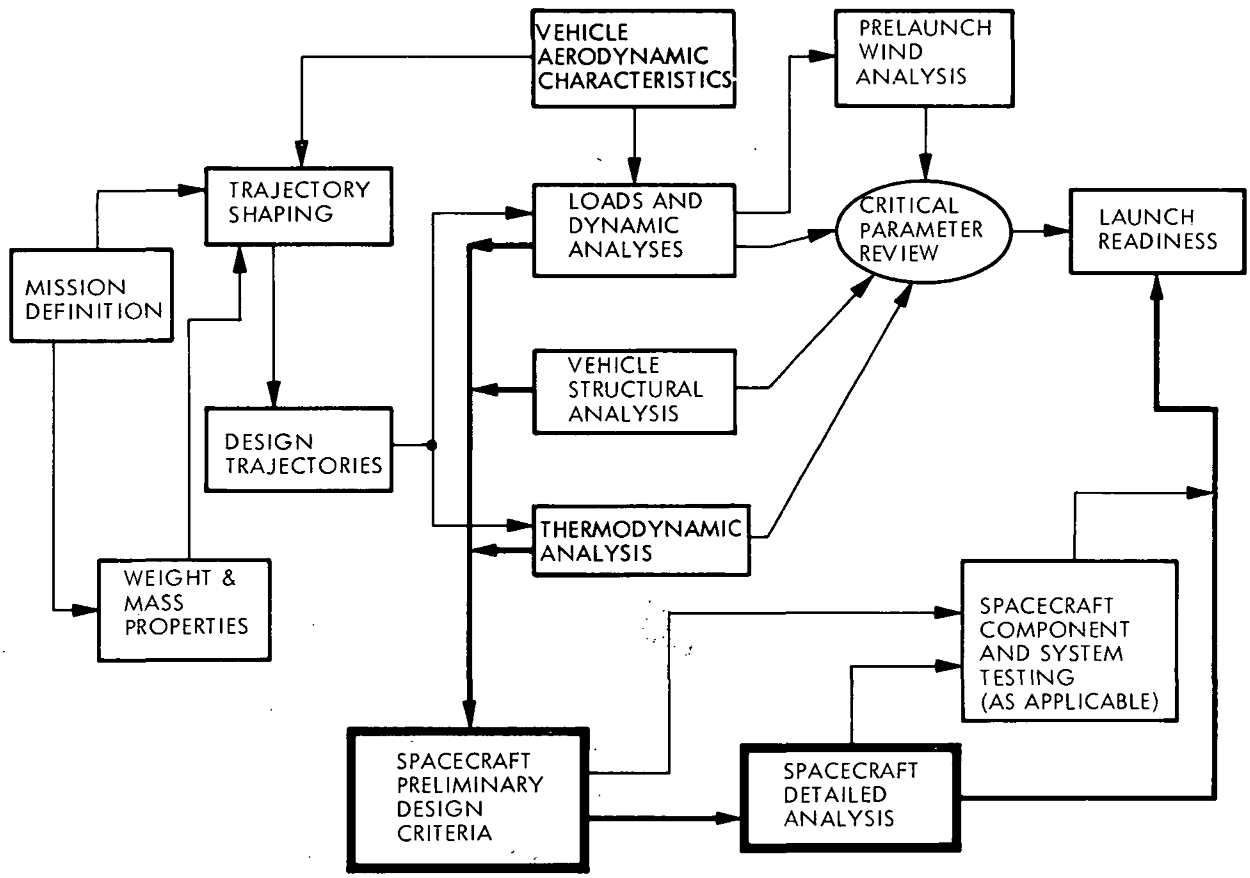


Fig. 4-1 Integrated Vehicle/Payload Analysis Flow

4.1 VEHICLE LAUNCH PROBABILITY

The launch probability of a vehicle may be defined as that percentage of time during which it may be launched under certain assumed or known ascent wind criteria without exceeding the established design constraints of the vehicle. Most Agena/booster/fairing systems have been designed for capability of ascent at ETR and WTR through upper atmosphere wind profiles having a 95 percent probability of not being exceeded. The vehicle maximum response due to ascent winds most usually occurs in the 20,000- to 40,000-ft altitude band.

The wind criteria most often used in establishing vehicle launch probability are listed below in order of their descending conservatism:

- Windiest-month concept
- Windiest-season concept
- Actual wind soundings

The windiest month and seasonal concepts are conservative synthetic profiles derived from accumulated actual wind soundings at the Eastern and Western Test Ranges and compiled into usable design formats. These two criteria are usually employed in the analysis of vehicles that must be operational, for reasons of meeting certain limited and predefined launch windows, at all times of the year. The actual wind soundings are used not only to determine year-round probabilities but also as the primary criteria in the analysis of systems whose launch cycle or frequency are fixed within a specific time span during the year.

Should launch probability requirements exceed structural and/or control constraints, the problem may be alleviated by consideration of one or more of the following:

- Reshaping the trajectory
- Employing an autopilot load-relief system
- Altering vehicle launch window
- Strengthening local structural

Normally, the above considerations would not be applied to any significant degree. For most configurations, launch probabilities of the order of 95 percent or greater are achieved.

The following tabulation presents a sampling of launch probabilities (employing the windiest-month criteria) established for previously flown or presently flying Agena programs (or proposed configurations).

Booster/Program or Fairing Type	Launch Probability (Percent)
Thorad-Three Solids/Agena	
Nimbus-B	95
OGO-F	95
Lightweight Clamshell (LCS) Fairing	95
Thorad-Six Solids/Agena	
Lightweight Clamshell Fairing	90
Thorad-Nine Solids/Agena	
Lightweight Clamshell Fairing	95
Atlas/Agena	
Mariner Venus-67	95
Lightweight Clamshell Fairing	85-95
Titan IIIB	
Lightweight Clamshell Fairing	85-95
Titan IIID	95

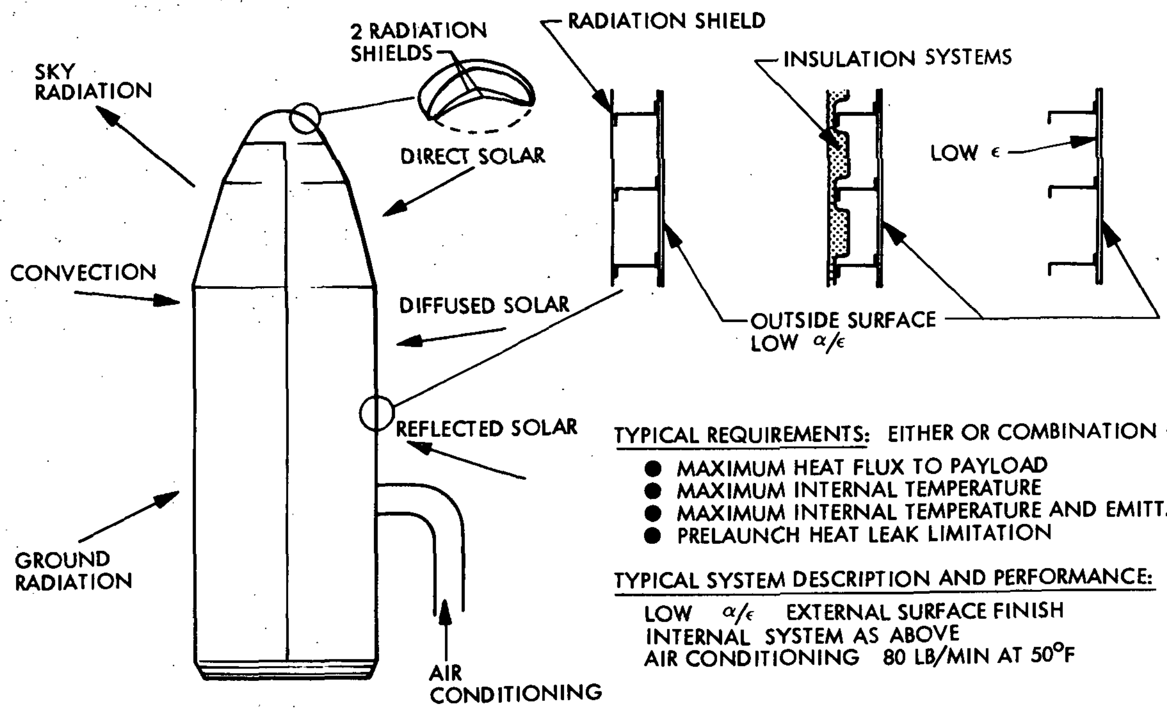
4.2 THERMAL HEATING AND THERMAL CONTROL

4.2.1 Spacecraft Prelaunch and Ascent Thermal Control

Thermal control of a spacecraft is usually provided on the launch pad to maintain acceptable temperature levels during prelaunch or to precondition or subcool the spacecraft prior to launch. Heating from spacecraft equipment operation or from external sources can be balanced by air conditioning, cooling blankets or sheaths, or other methods to prevent overheating. Air-conditioning provisions are available at each launch pad.

Design of the thermal control system results from considering both the prelaunch and the ascent requirements. Normally, the fairing internal thermal system is dictated by the spacecraft ascent requirements, such as a maximum heat flux or maximum temperature-emittance combination. Fairing thermal systems may include (1) only

polished, low emittance (ϵ) internal surfaces, (2) radiation shields, or (3) insulation systems. Prelaunch requirements are normally satisfied by direct air conditioning into the fairing at a rate of approximately 80 lb/min at 50°F. External fairing air-conditioning systems (cooling blankets and/or styrofoam sheaths) have been used previously when internal contamination or sterilization requirements did not allow direct air conditioning of the payload volume. Internal and external insulation systems may also be added to the fairing to limit heat leak rates. Figure 4-2 illustrates a typical fairing thermal control system, showing the on-pad external heat sources, typical requirements, and system description and performance.



TYPICAL REQUIREMENTS: EITHER OR COMBINATION OF

- MAXIMUM HEAT FLUX TO PAYLOAD
- MAXIMUM INTERNAL TEMPERATURE
- MAXIMUM INTERNAL TEMPERATURE AND EMITTANCE
- PRELAUNCH HEAT LEAK LIMITATION

TYPICAL SYSTEM DESCRIPTION AND PERFORMANCE:

LOW α/ϵ EXTERNAL SURFACE FINISH
INTERNAL SYSTEM AS ABOVE
AIR CONDITIONING 80 LB/MIN AT 50°F

Fig. 4-2 Spacecraft on-Pad Thermal Control

Figure 4-3 is a typical plot of total on-pad external heating as a function of the fairing average temperature. Higher α/ϵ ratio surface finish will shift the curve to the right, increasing the fairing average temperature. Heat transfer through the fairing and heat generated by the spacecraft are absorbed by the air conditioning.

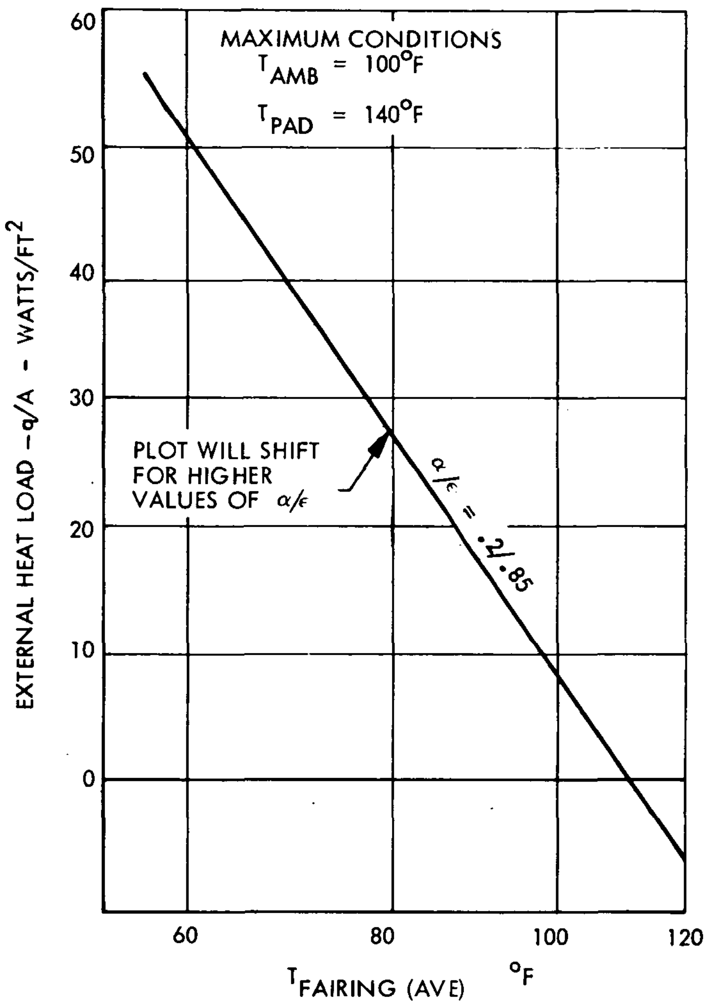


Fig. 4-3 Typical Fairing on-Pad Exterior Heating

While the spacecraft is enclosed in the fairing during ascent, the energy it dissipates must be accommodated by the heating of spacecraft or Agena masses, which may have been precooled. Once the fairing is jettisoned, the spacecraft is exposed to normal orbit heating environments as well as to free-molecular heating; the extent of the free-molecular heating depends on the altitude and surface orientation. "Rotisserie" or "toasting" attitude maneuvers may be employed as desired.

4.2.2 Aerodynamic Heating

The response of the spacecraft fairing structure to the aerodynamic heating associated with the boost-phase trajectory influences spacecraft design because it produces part of the spacecraft thermal environment and because it affects the clearance available between the fairing and the spacecraft.

In general, high-performance (payload pounds into orbit) trajectories result in high fairing temperature levels and significant circumferential temperature variations (resulting from pitch-plane maneuvers), with the downrange (+Z) axis being the hottest location. However, yaw-plane maneuvers (such as doglegs required to satisfy range safety constraints on low-inclination-orbit missions) result in moving the hottest circumferential position from the +Z axis.

The spacecraft fairing transfers heat energy to the payload by virtue of the temperature differences between the two structures. The mechanism for most of this boost-phase heat transfer is radiation; therefore, the fairing temperature level is usually minimized by use of high-emittance external surfaces to enhance external energy rejection from the fairing. Control of the degree of heat energy transfer from the fairing to the spacecraft is specified by the spacecraft designer. As mentioned in the preceding paragraph, this energy transfer is typically controlled by the use of low-emittance internal surfaces, radiation shields and insulation systems, or alternately, by constraining the trajectory.

The curves shown in Fig. 4-4 indicate the mechanism for limiting the effects of ascent heating by means of radiation shields. As shown, the outboard shield temperature profile reveals a time lag that limits its temperature to a fraction of the fairing skin temperature during the typical ascent period. Since the inboard shield is exposed to the temperature of the outboard shield, its temperature response lags even further. Therefore, the spacecraft is never exposed to the actual fairing skin temperature, since the fairing is ejected before the inner shield temperature increases to high levels.

Radiation shields having low emittance characteristics can also be installed immediately under the nose dome, as indicated in Fig. 4-2, in either single or multiple "coolie hat" configurations to protect the spacecraft from the heated nose dome during ascent.

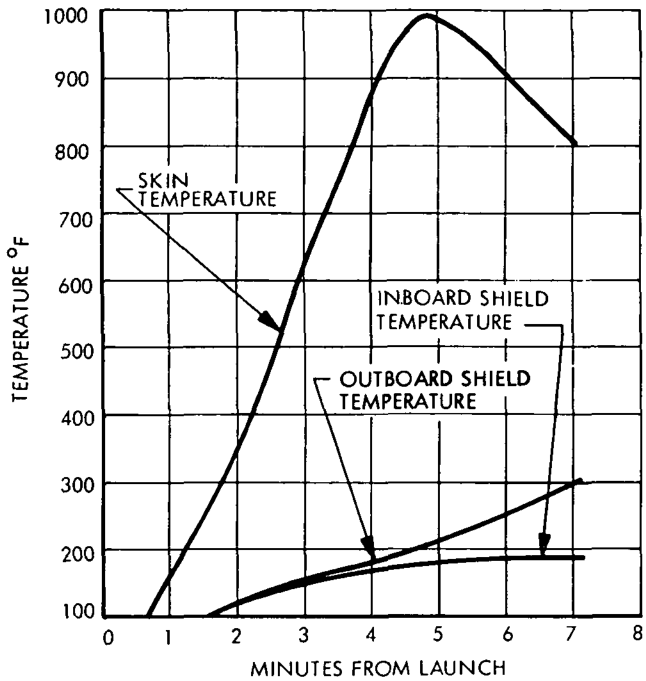


Fig. 4-4 Radiation Shield Temperature Profile

4.3 SPACECRAFT PRESSURE ENVIRONMENTS

Internal fairing pressures experienced during ascent flight do not normally present serious problems. This is because of the inherent openness of most spacecraft designs (which minimizes or effectively eliminates differential pressure loadings) and the fact that Agena fairings have flight-proven venting systems that minimize internal cavity pressures.

Spacecraft compartment pressures result from venting of the fairing cavity during ascent flight. During ascent, the air inside the fairing exhausts through the vent ports so that resulting internal pressures generally follow ambient pressure. A typical venting system is shown in Fig. 4-5.

A certain degree of retarding or accelerating of this airflow occurs as a consequence of the external aerodynamic flow field in the area of the fairing vent ports and points of external leakage. These aerodynamic perturbations are most prominent during the flight regime around Mach 1, when the aerodynamic pressure field undergoes rapid and drastic changes; i.e., shock mitigations along the vehicle. This retardation and acceleration of

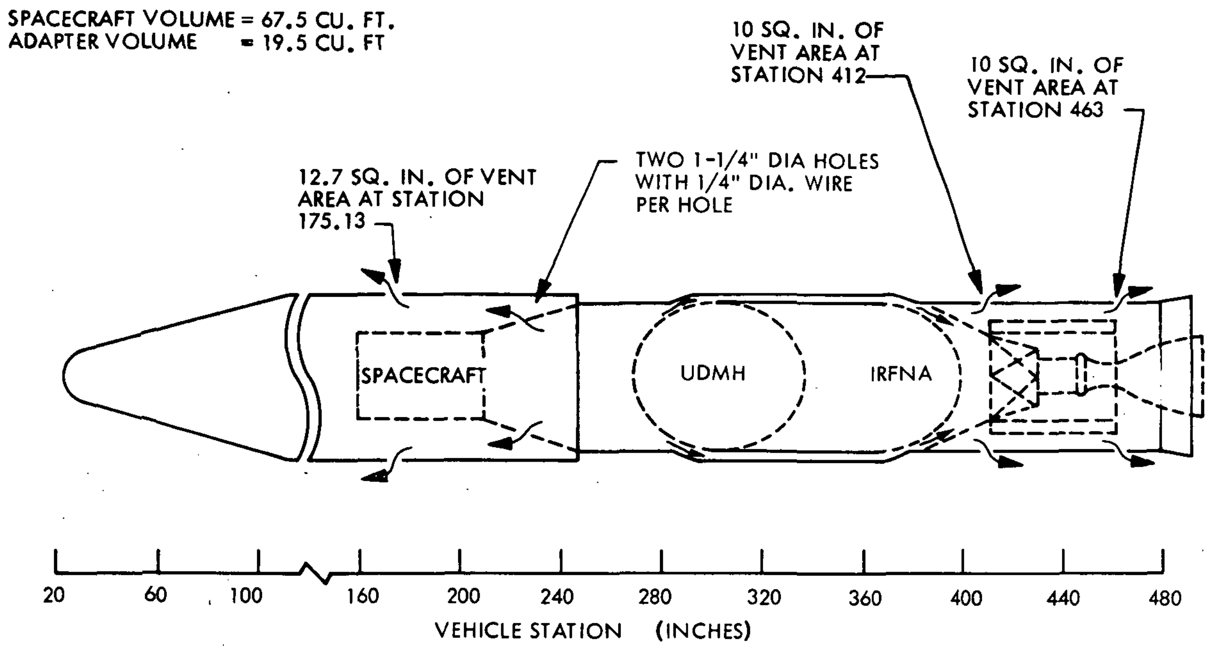


Fig. 4-5 Typical Venting Flow Diagram

the exhausting air from the spacecraft compartment respectively increases and decreases the internal pressure from ambient conditions. Deviations of these internal pressures which are too large or too rapid from ambient can result in structural loading and potential contamination conditions for the spacecraft. Care is therefore taken in the design of the fairing and vehicle venting systems to minimize these deviations within the spacecraft compartment.

The magnitude of the fairing internal pressures is a function of the ascent vehicle trajectory as well as the fairing configuration and location of the vent ports. Typical spacecraft compartment pressures for some previously flown Agena programs employing fairings with venting systems designed to minimize the previously described effects are indicated in Table 4-1

A spacecraft design will normally provide for venting the spacecraft to minimize differential pressures resulting from spacecraft compartment pressure variations. Unless spacecraft structures and equipment items are designed to withstand the induced differential pressure environment, there should not be any entrapped air cavities within the

spacecraft that cannot vent to the fairing cavity. For normal spacecraft designs, such as the wide family of spacecraft flown on previous booster/Agena combinations, these pressures and venting requirements do not impose significant design problems.

Table 4-1
TYPICAL SPACECRAFT COMPARTMENT PRESSURES

Booster/Spacecraft	Compartment Pressure Below Ambient $(P_i - P_\infty) < 0$	Compartment Pressure Above Ambient $(P_i - P_\infty) > 0$
Thorad/Nimbus-B	-1.05 psi	1.45 psi
Atlas/Mariner-Venus-67	-0.43 psi	1.12 psi
Titan IIIB	-0.04 psi	0.37 psi

4.4 FLIGHT DYNAMIC ENVIRONMENTS

Vibration and loading environments are described in the following pages for spacecraft used with an Agena. Spacecraft acoustic and random vibration environments have been based on either a 5-ft-diameter metallic fairing (LCS fairing configuration) or a 10-ft-diameter metallic fairing configuration. Other fairing configurations affect the magnitude of these environments; Lockheed may be consulted to determine the nature of these changes if a different fairing configuration is desired. The assumption is also made that an adapter system mounts the spacecraft to the Agena. If a spacecraft design is employed that "close couples" to the Agena station 247 interface ring (when used in conjunction with the LCS fairing configuration), the environments may be affected. Lockheed may be consulted to determine the effect of this type of installation.

Load factors and stiffness characteristics for various booster/Agena systems are available as a guide for the design of the total spacecraft structure, major structural elements, and equipment mounting structure. Acoustic, random vibration, transient vibration, and shock environments are factors to be considered in the design. The ability of spacecraft structures to survive these environments is historically demonstrated by test. These environmental data may also be used in the design of equipment, boxes, major experiments, etc. The equipment or experiment designer may consider acoustic, random

vibration, transient vibrations, sinusoidal vibrations, and total accelerations in the design of specific items of equipment, selecting components and size elements to withstand these flight environments. The proof of design adequacy is normally demonstrated by testing to environments.

4.4.1 Acoustic

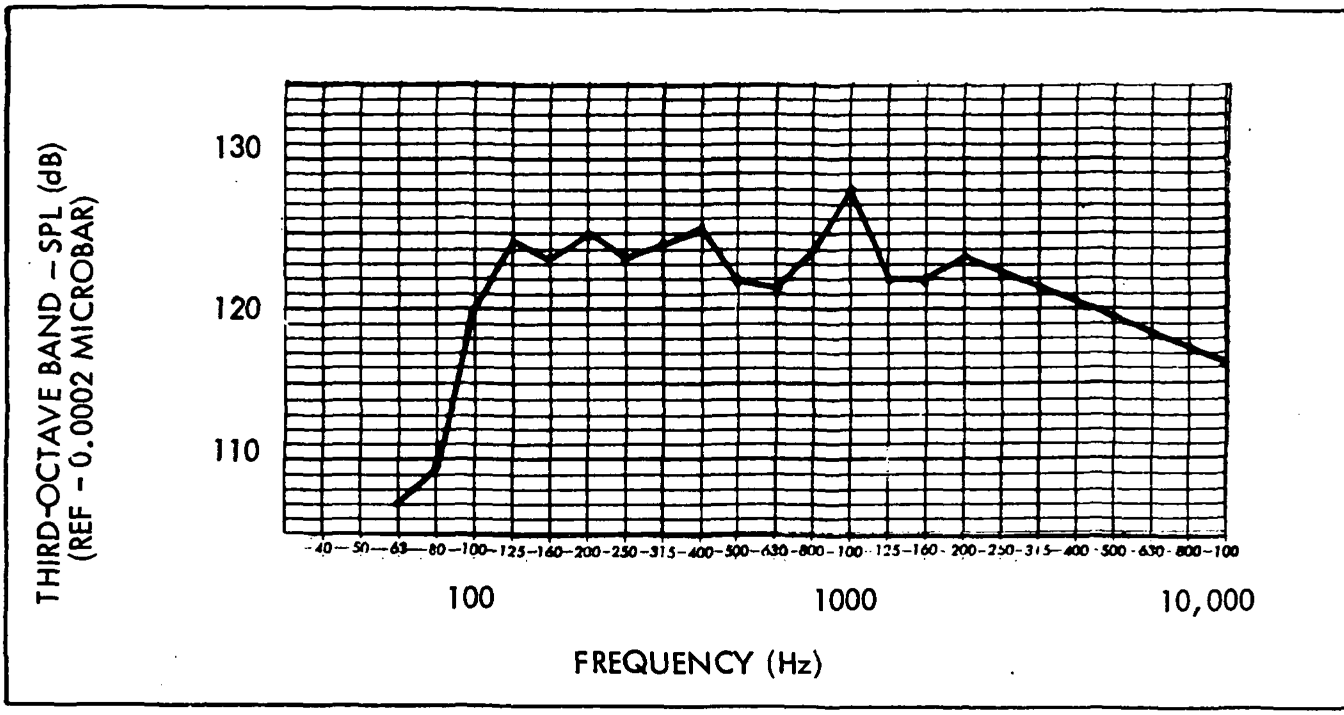
The maximum acoustic environment occurs during liftoff and transonic flight. The liftoff acoustic environment originates from ground reflections of the acoustic field generated by the rocket exhaust stream and lasts for approximately 5 sec. The field is most intense near the engines and decreases toward the nose of the launch vehicle. The transonic acoustic environment originates from pressure fluctuations on the external skin and is transmitted mechanically through the structure as well as acoustically through the internal atmosphere within the payload fairing and within the spacecraft. The transonic acoustic environment lasts for approximately 55 sec during the period of transonic flight and maximum dynamic pressure. Examples of typical spacecraft acoustic environments on a Titan/Agena system, which are composites of the liftoff and transonic spectra, are presented in Figs. 4-6 and 4-7. These environments are present for a total duration of 1 minute.

4.4.2 Random Vibration

Random vibration is produced during first-stage flight primarily by the liftoff acoustic field (approximately 5-sec duration) and by boundary layer turbulence acting on the external vehicle skin during transonic flight and during the time of maximum dynamic pressure (approximately 55-sec duration).

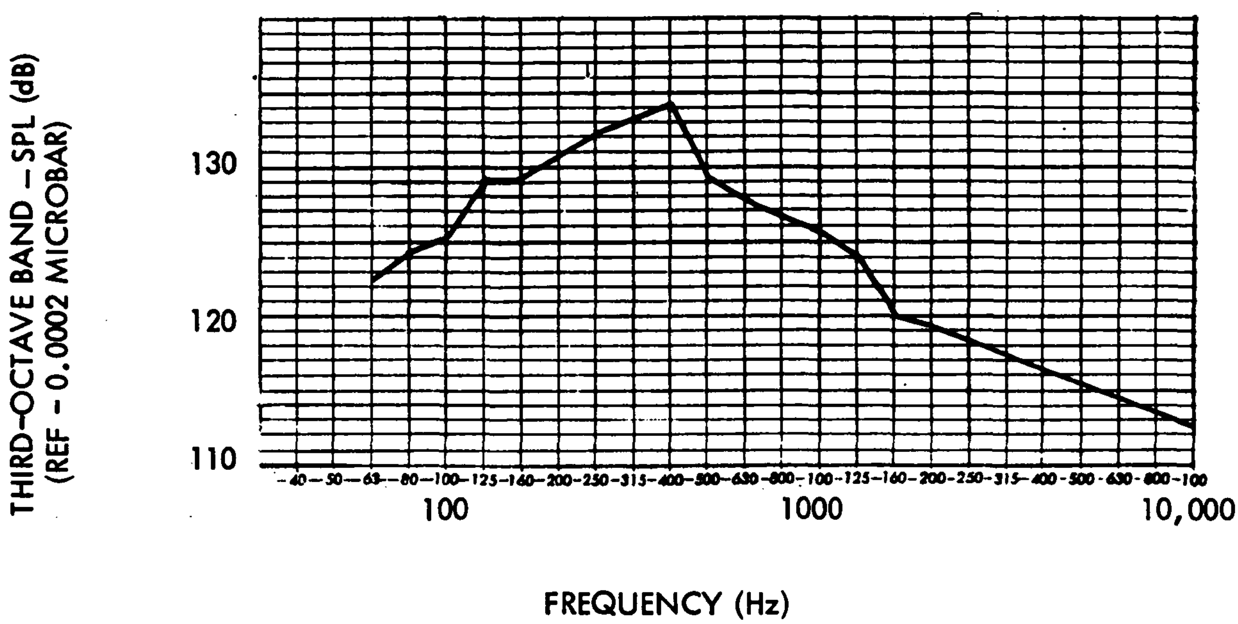
A typical random vibration profile for a spacecraft mounted equipment installation during the periods encompassing liftoff and transonic flight is shown in Fig. 4-8. A corresponding histogram, indicating the cumulative time duration for which any given percentage of the peak random vibration levels exist, is also shown in this figure.

Figure 4-9 is a graph of random vibration environments produced during periods of Agena engine burn. Since there is no sensible atmosphere during this engine burn,



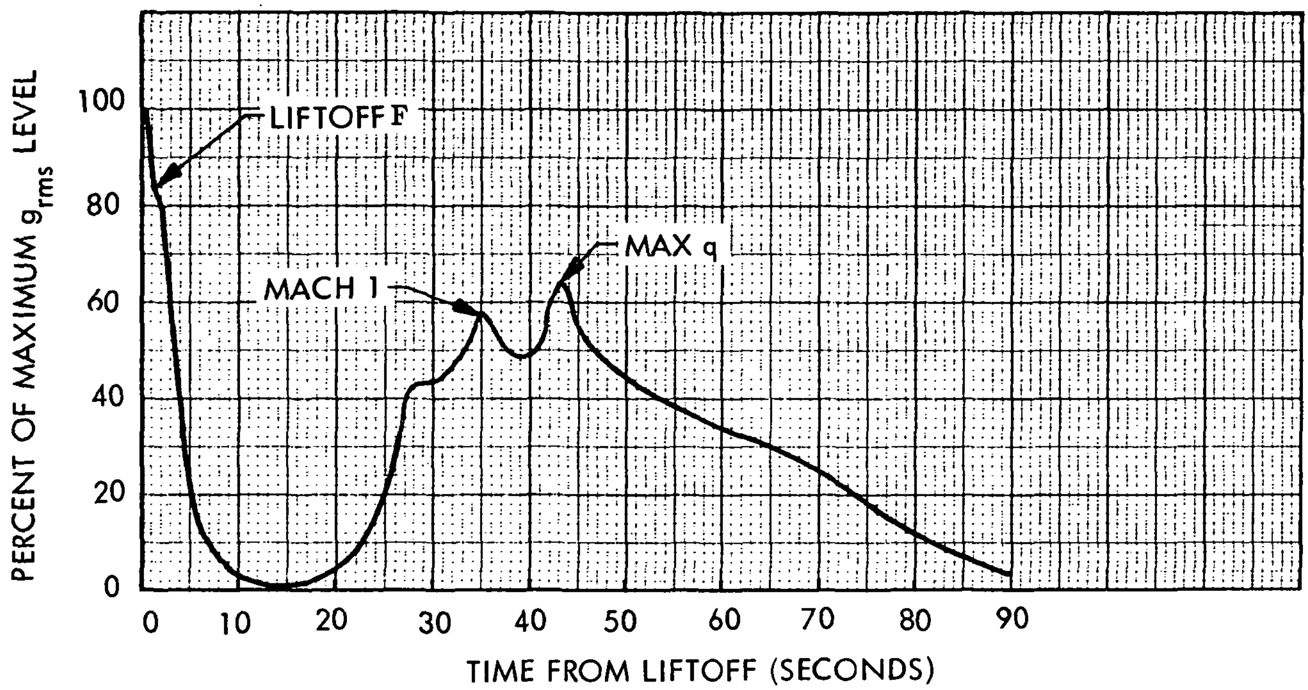
FREQUENCY (Hz)	SOUND PRESSURE LEVEL (dB)	FREQUENCY (Hz)	SOUND PRESSURE LEVEL (dB)
63	107.0	1000	128.0
80	109.5	1250	122.0
100	120.0	1600	122.0
125	124.5	2000	123.5
160	123.5	2500	122.5
200	125.0	3150	121.5
250	123.5	4000	120.5
315	124.5	5000	119.5
400	125.5	6300	118.5
500	122.0	8000	117.5
630	121.5	10,000	116.5
800	124.0	OVERALL	136.0

Fig. 4-6 Typical Spacecraft Acoustic Environment – Titan IIIB/Agena (LCS Fairing)

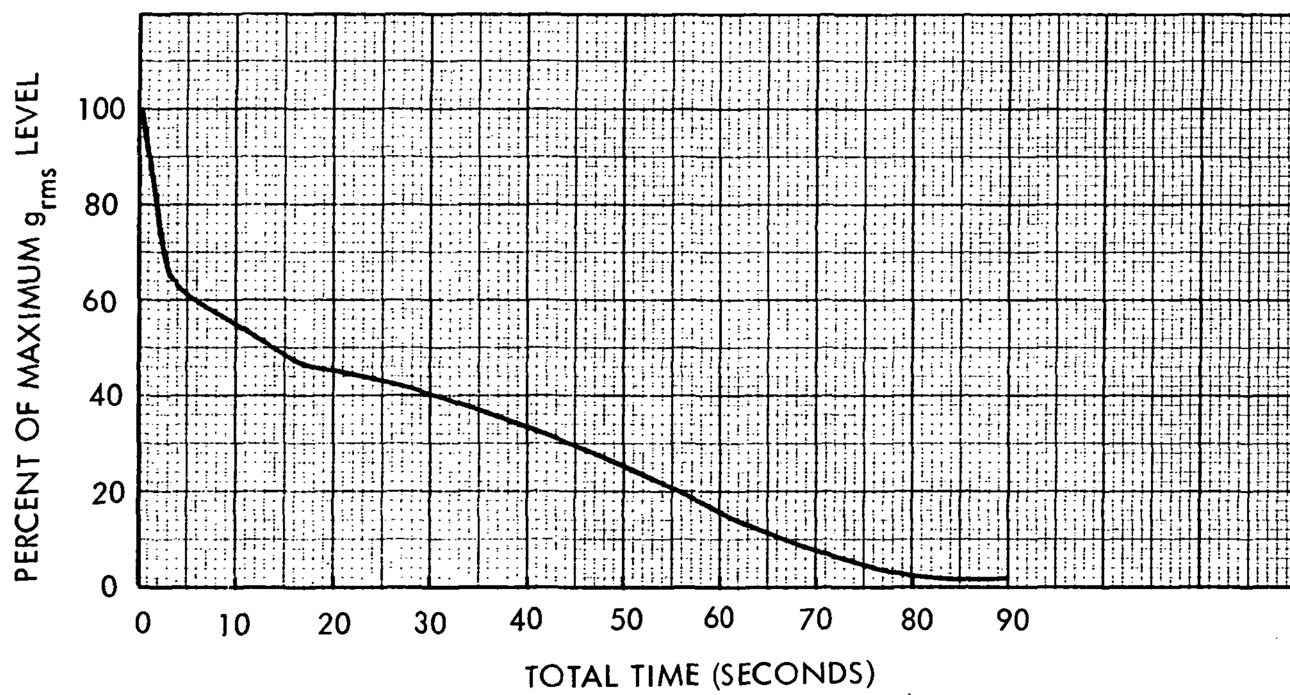


FREQUENCY (Hz)	SOUND PRESSURE LEVEL (dB)	FREQUENCY (Hz)	SOUND PRESSURE LEVEL (dB)
63	122.0	1000	125.5
80	124.0	1250	124.0
100	125.0	1600	120.0
125	129.0	2000	119.5
160	129.0	2500	118.5
200	130.5	3150	117.5
250	132.0	4000	116.5
315	133.0	5000	115.5
400	134.0	6300	114.5
500	129.0	8000	113.5
630	127.5	10,000	112.5
800	126.5	OVERALL	141.0

Fig. 4-7 Typical Spacecraft Acoustic Environment – Titan IIIB/Agena (10-Ft-Diameter Fairing)



TYPICAL RANDOM VIBRATION FLIGHT PROFILE



TIME FOR WHICH VARIOUS g_{rms} LEVELS ARE EXCEEDED

Fig. 4-8 Typical Ascent Phase Random Vibration Profile

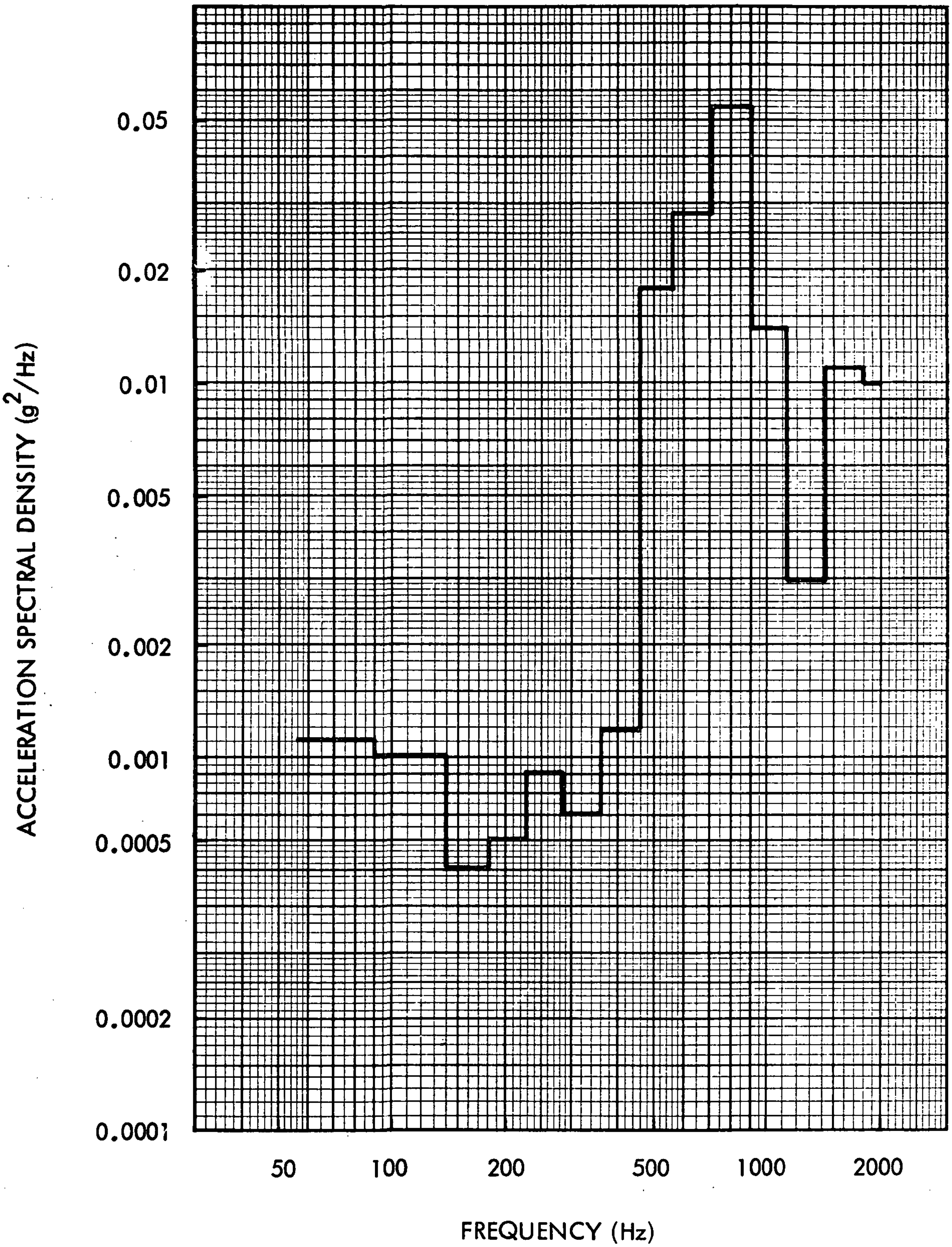


Fig. 4-9 Spacecraft Interface Engine-Induced Random Vibration Environment

acoustic fields are not generated; the environment present is solely that produced by structural transmission of the engine vibrations. This environment exists at an essentially constant level for a total period of approximately 5 minutes. The Agena random vibration environment was derived by enveloping a number of measurements made on different types of equipment installations near the spacecraft interface.

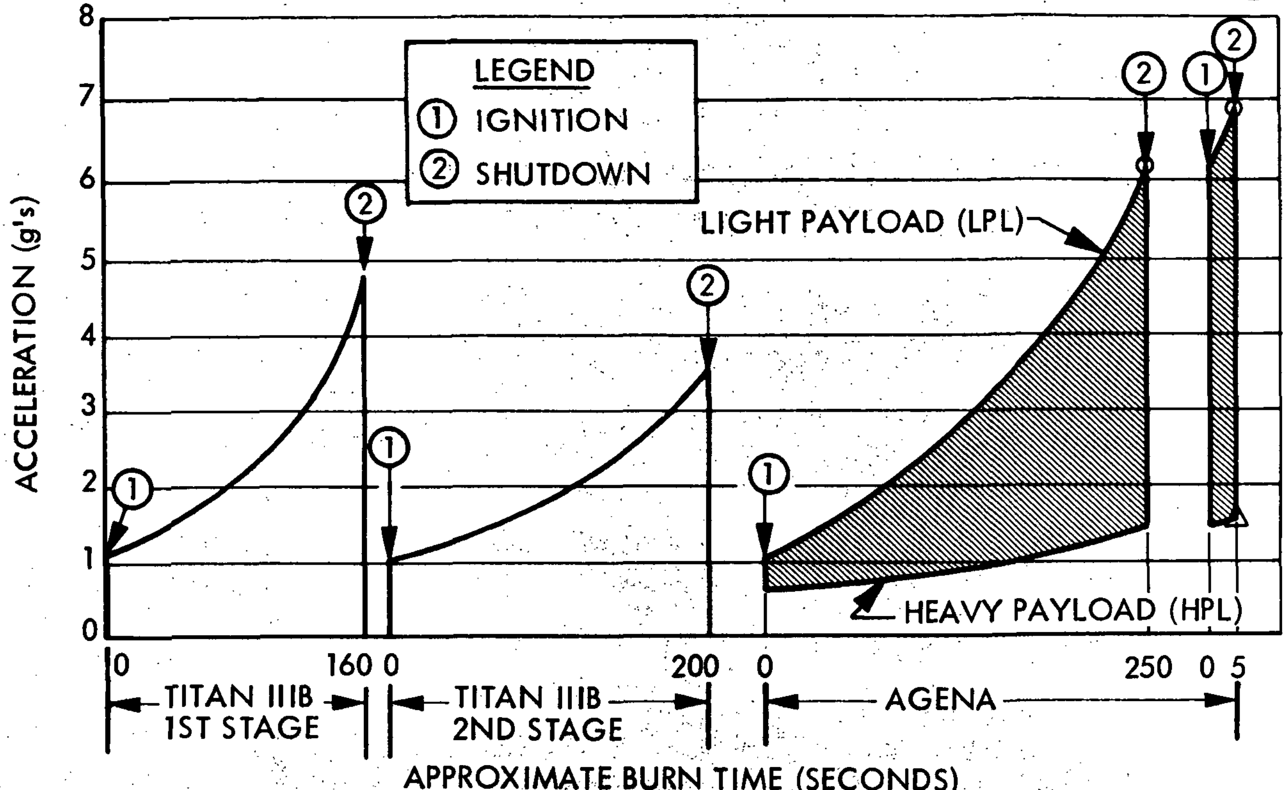
4.4.3 Transient Vibration

On a typical Titan/Agena system, transient vibrations in the approximate frequency range of 15 to 400 Hz occur during the various boost-stage engine ignition and shutdown events, as well as during gust and buffeting events. The Titan first-stage shutdown and the Agena ignition and shutdown events produce the more severe transients. Transient vibration environments for spacecraft structure and equipment are described by means of shock spectra that are a measure of the response of a single-degree-of-freedom spring-mass system to the transient input. The spacecraft shock spectrum is based on an effective amplification factor of 1.5 between 25 and 100 Hz in the longitudinal axis and 3.0 between 8 and 15 Hz and between 25 and 60 Hz in the lateral axes to account for spacecraft modal amplification. The shock spectrum for spacecraft equipment is based upon a Q of 10 (effective amplification factor of 3.7) between basic structure and the equipment based in the frequency range from 30 to 200 Hz. (Q is a means of expressing the critical damping factor ζ, where $Q = \frac{1}{2\zeta}$.)

Pure sinusoidal environments do not exist to any appreciable degree for Titan IIIB/Agena vehicles. To the degree that this environment may exist, it is considered to be adequately covered by the transient environment presented.

4.4.4 Steady-State Acceleration

The spacecraft and its equipment experience steady-state acceleration due to engine thrust. Maximum steady-state acceleration occurs at the various stage burnout events, as indicated in Fig. 4-10. Spacecraft weight becomes a significant factor in the acceleration time history after booster/Agena separation. A typical Agena acceleration time history for a minimum and maximum spacecraft weight is presented in Fig. 4-10.



SUMMARY OF STEADY STATE ACCELERATIONS (g's)

BOOST EVENT	TITAN IIIB		AGENA			
	1ST STAGE	2ND STAGE	1ST BURN		FINAL BURN	
			LPL	HPL	LPL	HPL
IGNITION	1.2	1.1	1.02	0.67	6.2	1.44
SHUTDOWN	4.75	3.5	6.2	1.44	6.9	1.47

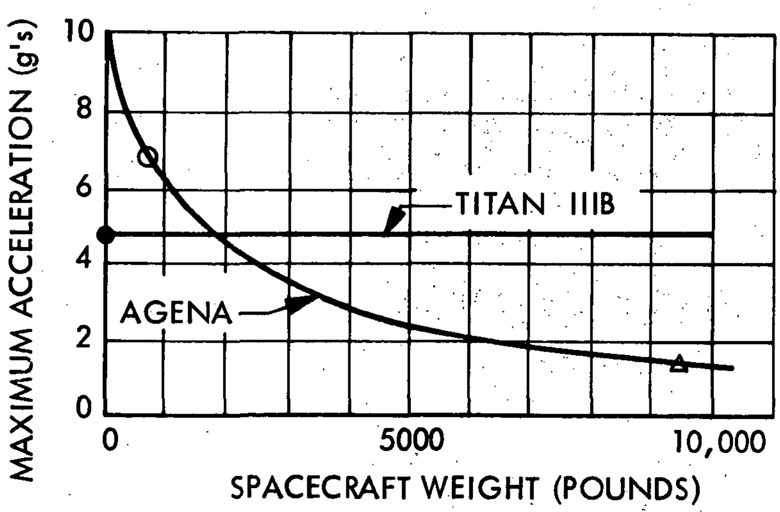


Fig. 4-10 Acceleration Histories as a Function of Flight Time and Spacecraft Weight – Titan IIIB/Agena

Agena acceleration time histories for spacecraft of intermediate weight may be obtained by use of the attendant weight-reduction curves. For the Agena burn,the spacecraft consists of the total vehicle forward of Agena Station 247.

4.4.5 Pyrotechnic Shock

Pyrotechnic shock originates from pyro devices, such as stage and fairing circular separation joints, spacecraft separation bolts, pinpullers, and captive nuts. The most severe shock levels experienced by spacecraft equipment are due to fairing separation at the Agena interface (Agena Station 247), spacecraft separation, and activation of spacecraft pyro devices such as pinpullers and captive nuts.

Pyrotechnic shock is not intended to be interpreted as the loading condition for design of prime equipment package structure or subassembly structure. However, sensitive equipment may experience the pyrotechnic shock environment. The designer may locate susceptible equipment a safe distance from the pyro shock source and/or provide for some means of shock isolation, if necessary. Equipment isolators are available that can reduce shock levels by as much as a factor of 10. Typical types of equipment sensitivities to pyrotechnic shock environments that have been experienced during previous pyro shock testing are relay chatter, meter pulse rate shift, inferior bond and weld fracture, and crystal failure. Suspected sensitive equipment should be qualified for the pyro shock environment.

If pyrotechnics are not used within the spacecraft itself or to separate the spacecraft from the Agena, the fairing separation and the Agena/booster separation could be the only sources of pyrotechnic shock environment to which the spacecraft would be subjected. This environment, while considerably attenuated by the time it reaches the spacecraft/Agena interface, could be of significance to sensitive elements on the spacecraft.

4.4.6 Design Load Factors and Frequency Constraints

Maximum dynamic accelerations are experienced by spacecraft and spacecraft-mounted equipment when the spacecraft structure couples with fundamental vehicle modes.

Events that produce the most severe dynamic accelerations are engine ignitions and shutdowns. These dynamic excitations have been appropriately combined with steady-state thrust loading to determine the maximum loading conditions and are available as design load guidelines for spacecraft structure and for equipment mounting bracketry.

The Agena longitudinal load factors are available as a function of spacecraft weight; therefore, given the spacecraft weight, the maximum longitudinal design limit load factor may be found.

Significant coupling with the basic vehicle fundamental modes must be avoided. The spacecraft should be designed so that fundamental frequencies are above the vehicle fundamental frequencies for both longitudinal (thrust) and lateral axes. These frequency constraints are in reference to the cantilever modes of the spacecraft; that is, hard-mounted at the spacecraft separation plane. Spacecraft with lower fundamental frequencies can be accommodated, however, if the effect of these lower frequencies on the design load factors is assessed.

4.5 DYNAMIC TEST REQUIREMENTS

Test requirements and test programs for spacecraft and spacecraft equipment are determined by the spacecraft designer or spacecraft agency. Accordingly, the test data and test methods described here, although extensive and relatively complete, are not mandatory requirements. They may, however, be used as guides for the formulation of a test program. Many items of spacecraft equipment may have been either previously flown or have been tested to other requirements and can be qualified by similarity.

Qualification test levels have generally been established by increasing the expected flight levels by a factor of 1.25. The spacecraft designer may accept this factor or may determine the degree of test margin desired to achieve qualification of a particular spacecraft structure for each test environment. Qualification and acceptance test criteria are presented first for the total spacecraft system and structure, then for items of equipment or experiments mounted on the basic spacecraft structure. Recommended test specification tolerances are presented in Table 4-2.

Table 4-2

TEST SPECIFICATION TOLERANCES

Random Vibration	±2 dB on g^2/Hz (Factor of 1.59) ±1 dB on overall g_{rms} (Factor of 1.12)
Sinusoidal Vibration	±10% on g's (0-peak) ± 2% or 1 Hz on frequency, whichever is greater
Mechanical Shock	±10% on g's or seconds
Acoustic Spectra	±2 dB on overall sound pressure level ±3 dB on one-third octave band sound pressure level (minus tolerance waived in 50 to 80 Hz and 3150 to 10,000 Hz one-third octave bands to best facility capability)
Shock Synthesis Spectra	±2 dB on g's (0-peak) (Factor of 1.26)
Acceleration	±5 percent (at reference point)
Pyrotechnic Shock	±10 percent on g's x Hz

4.5.1 Total Spacecraft Tests

Spacecraft qualification tests verify the design adequacy of the primary and secondary structure, verify proper performance of critical spacecraft equipment when operated as part of the total system and, by monitoring response at equipment mounting locations, verify the adequacy of the specified equipment qualification test levels. For qualification testing, the test spacecraft assembly should be complete with functional and/or mass and c.g. simulated equipment. Systems level performance testing conducted during and after each dynamic qualification test will usually determine compliance with the system performance specification.

4.5.1.1 Acoustic Noise. Full-scale acoustic tests are often recommended as demonstrations of the ability of a spacecraft to survive induced acoustic environments. The recommended test spectra are derived by smoothing the flight acoustic spectra.

4.5.1.2 Random Vibration. Random vibration qualification testing of spacecraft at the systems level is generally not recommended because of the difficulty of obtaining a proper simulation of the vibration environment throughout the structure and because the primary source of flight random vibration environments results from the external acoustic fields. However, in some cases a random vibration test may be considered necessary at the systems level to demonstrate the ability of the spacecraft to survive the random vibration environment induced by the Agena engine burn.

4.5.1.3 Shock Synthesis. Shock synthesis testing is an appropriate and not unduly conservative means of simulating the transient flight environment resulting from vehicle response to external forces (i.e., engine ignition or shutdown). The principle of shock synthesis testing is to generate a complex waveform similar to that of the real transients encountered in flight. The damage potential of the synthesized waveform and the flight transients are assumed similar when the shock spectra, (i.e., the peak response levels of each) are equal. An electrodynamic shaker system may be used to synthesize the waveforms.

Prior to full-level qualification, it is suggested that a sine resonance search be conducted over a specified frequency range and amplitude and that the response be measured at suspected critical equipment/bracket interfaces. This test is recommended to ensure that the response of these installations is not such that the load factors that designed the brackets will be exceeded during full-level shock transient testing. If necessary, the shock transient test may be conducted at a reduced level prior to full-level testing to preclude the possibility of an unrealistic test failure. The test should be controlled at the spacecraft c.g.

4.5.1.4 Acceleration. Normally, acceleration testing is not used for qualifying a spacecraft; however, under certain conditions it may be necessary to subject a spacecraft to a centrifuge test. Specific examples: to simulate fluid loading on a tank or

tank skirt due to ascent steady-state thrust or to verify performance of a spacecraft subassembly that could be sensitive to steady-state acceleration, such as a gyro, when operated as a part of the total vehicle system.

4.5.1.5 Static Testing. Qualification testing for the maximum combined loading condition is normally achieved by means of static testing. Design load factors and pressure and thermal environments, as applicable, all increased by a proper factor to achieve the desired test margins, establish the loading environment.

4.5.1.6 Pyrotechnic Shock. The spacecraft module, complete with functional or mass and c.g. simulated equipment, may be subjected to pyrotechnic shock to qualify individual equipment items or to verify end-to-end performance of the spacecraft in a pyrotechnic shock environment. Pyrotechnic shock will not in itself produce a design loading condition on the spacecraft structure.

4.5.2 Equipment Tests

Qualification and acceptance test criteria are suggested in the following paragraphs for equipment or experiments mounted on spacecraft structures. Equipment qualification tests are designed to demonstrate the equipment's capability to survive the most severe flight conditions and to verify integrity of design. Equipment should be operated during testing if it is required to operate during the time of flight in which the environment that produced the test is present. Acceptance testing is conducted to demonstrate integrity of equipment assembly workmanship. Requirements for operating equipment during acceptance testing and for performing functional checks must be established by the spacecraft designer.

4.5.2.1 Acoustic Noise. Acoustic qualification testing of spacecraft equipment is normally not required since the random vibration test generally produces more severe component response. Some equipment, however, may be susceptible to damage from acoustic noise. Examples of the type of equipment and equipment components that can be excited by acoustic noise are those that have large surface areas and/or low-mass-to-area ratios with very low damping: sensitive components attached to thin-plate surfaces

(relays, piezoelectric devices, etc.); equipment with exposed diaphragm elements (valves, relays, switches, pressure-sensitive transducers, etc.); thin fiberglass antenna ground planes; and thin-plate radiators. Testing of such equipment should be considered.

Normally, acoustic acceptance testing of equipment is not conducted; however, some equipment, such as a thin-plate radiator or a solar array, may have properties such that the entire equipment item is most realistically excited by acoustic excitation.

4.5.2.2 Random Vibration. Random vibration testing to qualify payload equipment for use on boosted vehicles is recommended. Random vibration testing is generally recommended as the most appropriate single environment to be used for acceptance testing of equipment.

4.5.2.3 Acceleration. Acceleration testing is designed to test equipment to the worst-case combination of steady-state and dynamic loading. In the longitudinal axis of Titan IIIB/Agena vehicles, this condition occurs either at Titan stage shutdown or at Agena engine shutdown. Lateral acceleration values are based on predicted worst-case ground handling and ascent steady-state loading conditions.

Accelerations should be applied for a period of 5 minutes. For equipment with unspecified orientation in the spacecraft at the time of test or which may be reoriented on a later flight, the recommended acceleration may be applied in all axes to cover the contingency.

4.5.2.4 Transient Vibration/Mechanical Shock. Equipment may be qualified for the transient vibration environment by shock synthesis or mechanical shock testing. Shock synthesis testing is the recommended test method because mechanical shock testing produces an overtest on components with low Q's (Q is related to the critical damping factor ζ, $Q = \frac{1}{2\zeta}$), whereas shock synthesis testing closely reproduces the flight environment for any Q component. It is recommended that the shock synthesis test be repeated three times in each axis. If mechanical shock testing is used for simulation, it is recommended that the tests be repeated three times in each direction of each axis.

4.5.2.5 Pyrotechnic Shock. Equipment sensitive to pyrotechnic shock environments, such as equipment containing relays, crystals, and bonded parts, should be considered for pyro shock tests. Sensitive mechanisms, such as mechanically pulsed velocity meters, can be damaged by pyro shock if they are located near the source and not properly isolated. Pyrotechnic shock qualification of spacecraft equipment may be demonstrated by testing at the total spacecraft level or by testing the individual items of equipment.

4.6 SEPARATION EVENTS/ENVIRONMENTS

4.6.1 Booster/Agena Separation

Booster/Agena separation occurs during a coast period in an essentially zero-g environment immediately after booster engine shutdown. Sufficient time lapse is allowed for the booster engine tailoff thrust to reach a level that will be overcome by the retrorocket forces. This assures that the booster will not overtake and collide with the Agena after separation. During separation, no forces (engine thrust, etc.) are applied to the Agena. When the separation command is given, the circumferential joint connecting the booster adapter to the Agena is severed pyrotechnically, simultaneously with ignition of the retrorockets. The booster then moves slowly away from the Agena, with the separation guided by a roller/rail system between the booster adapter and the Agena aft rack structure. This separation event is a relatively slow, controlled event and results in negligible environments (pitch, yaw, and roll rates) being imparted to the Agena vehicle. The environments so produced are not generally of concern in the design of spacecraft systems.

Severing the Agena/booster adapter connection produces a high-frequency, short-duration, high-amplitude shock environment that is primarily limited to the Agena aft rack and engine cone structures. This pyrotechnically induced shock environment attenuates rapidly with distance and, with regard to spacecraft equipment and structure, can be considered less severe than the shock environments produced by the fairing separation or separation of the spacecraft itself.

4.6.2 Agena/Spacecraft Fairing Separation

The fairing separation method depends on the fairing configuration employed, either over-the-nose or clamshell. The former type is separated during a coast period in which the vehicle is in an essentially zero-g environment; the latter type is separated either during the latter stages of first-stage boost or during the initial stages of Agena burn. The fairing separation event is programmed as early in the ascent trajectory as practical to enhance overall system performance. The fairing is generally separated, within the constraints mentioned above, as soon as the vehicle has reached an altitude where the aerodynamic forces are essentially zero and where the free-molecular heat rates, which would be imparted to the spacecraft, are negligible.

Fairing separation does not induce significant loads or impart rates to the Agena vehicle that would be of concern for spacecraft design. Both types of fairings are separated by helical compression spring devices at the base of the fairing. The fairing/Agena circumferential interface joint is severed pyrotechnically; for clamshell fairings, the longitudinal joint is also separated by a pyrotechnic joint. Explosive debris and contamination are controlled by a special design of the separation joint.

The high-frequency, short-duration, high-amplitude shock environment at the fairing/Agena interface induced by fairing pyrotechnic separation attenuates rapidly as distance from the source increases.

4.6.3 Spacecraft Separation

4.6.3.1 Spin-Stabilized Mission. On three-stage missions employing a kick stage, a spin-stabilized system is used. Prior to third-stage ignition, a spin table, a third-stage motor, an adapter fitting, and the spacecraft are spun up to the desired spin rate by the firing of spin rockets mounted on the spin table. Rockets of different thrust characteristics can be used in pairs in various symmetrical combinations to achieve the desired spin rate. Generally, spin rates of between 40 and 100 rpm can be achieved for spacecraft having roll moments of inertia up to 60 slug ft^2. For spacecraft with greater moments of inertia, the upper limit of the spin rate capability decreases as

inertia increases. Spin rates of less than 40 rpm can be achieved; however, effect on orbit injection errors must be evaluated to assess the practicality of lower spin rates.

Third-stage spin-up (positive roll direction is clockwise, looking forward) imposes angular acceleration loads on the spacecraft. The magnitude of this acceleration depends on the total roll moment of inertia of the spinning mass and the thrust profile of the spin rocket array. Angular acceleration between 6 and 20 radians/sec^2 can be expected when the TE364 motor is used and between 10 and 50 radians/sec^2 when the FW-4 motor is used as the third stage. Peak angular acceleration will occur when the spin rockets achieve maximum thrust and will have a nominal duration of 30 msec.

The spacecraft is fastened to the third stage attach fitting by a two-piece V-band-type clamp secured by two bolts. Spacecraft separation is achieved by actuation of ordnance cutters that sever the two bolts. Relative separation velocities of approximately 6 to 8 ft/sec are imparted to the spacecraft by a separation spring (or springs), depending on the attach fitting configuration. A rocket or yo-yo tumble system is usually incorporated that tumbles the expended third-stage motor 2 sec after spacecraft separation.

4.6.3.2 Non-Spin-Stabilized Mission. For missions not employing a third-stage kick motor, spin stabilization of the spacecraft is generally not required. Spin stabilization can be provided, however, as on the NASA ISIS-X mission, in which a spin table adapter system mated directly to the Agena was employed.

Generally, separation of spacecraft not requiring spin stabilization is achieved during a coast period immediately after Agena engine shutdown, when the vehicle longitudinal acceleration is equal to zero. The separation system for a typical spacecraft (Lunar Orbiter: weight, 845 lb; pitch and yaw inertias, 110 slug-ft^2) consists of a V-band release and four spring mechanisms. The V-band is separated into two parts by pyrotechnic cutters. The helical compression springs then provide the energy to separate the spacecraft from the Agena. Relative separation velocities depend on separation spring sizes and forces and on the mass and inertia of the spacecraft. For the Lunar Orbiter spacecraft, a relative separation velocity of 1.6 ft/sec was achieved with a maximum rotational rate of 2.3 deg/sec, induced by separation springs having a spring rate of 28 lb/in. acting over a 1.5-in. stroke. Other standard separation systems can

be used for particular spacecraft adapter systems. Separation spring size and tolerance requirements are specified to meet specific spacecraft separation requirements and are normally verified by a spacecraft separation system analysis and test.

4.6.4 Balancing

If spin stabilization is used, with or without third-stage kick motors, precise balancing of the spacecraft (and the motor, as applicable) may be required. Spin balance requirements are dictated by the desired injection accuracy and attitude. The balancing operation may require the addition of masses to achieve an acceptable balance. Static and dynamic balancing may be required; each mission must be evaluated individually to determine whether either or both balancing modes are necessary.

4.6.4.1 Static Balance. For spinning spacecraft to be statically balanced, the spacecraft c.g., must not be displaced greater than 0.015 in. from the spacecraft centerline. The centerline is defined as a line perpendicular to the mating surface and passing through the theoretical center of the spacecraft/attach fitting diameter.

For nonspinning spacecraft in which there is a separation tipoff restraint, the c.g. should be held as close as practical to the centerline of the spacecraft. The location of the spacecraft c.g. offset should be identified to within 0.03 in. In cases where it is impractical for reasons of design to maintain the c.g. extremely close to the spacecraft centerline and there is still a constraint to minimize spacecraft angular rates at separation, spring ejection systems can be designed to compensate for the c.g. offset and still maintain the desired accuracy on tipoff rates.

4.6.4.2 Dynamic Balance. The product of inertia (dynamic unbalance) causes nutation; i.e., a collapsing and expanding of the wobble angle about that wobble angle caused solely by the tipoff rate. The degree of nutation is dependent upon the ratio of roll inertia to the pitch and yaw inertias. To be dynamically balanced, the angle between a principal axis of inertia of the spacecraft and its intended spin axis (spacecraft centerline) should not be greater than 0.002 radian.

4.6.5 Spacecraft Alignment Restraints

The spacecraft should incorporate a means on or near the top to measure runout (total indicator reading) of the rotating fully assembled third stage and spacecraft to verify alignment.

Section 5
SOFTWARE PROGRAMS

Section 5
SOFTWARE PROGRAMS

In the Ascent Agena guidance system, an airborne digital computer is used to perform mathematical calculations for navigation and guidance and to steer and control all stages of the launch vehicle during the ascent phase. To provide the verified intelligence to the flight hardware, a tested flight program is loaded into the airborne computer during prelaunch activities. These paragraphs describe the flight program, a prelaunch system test program, and the comprehensive process used to validate the instructions and information prior to airborne computer loading.*

5.1 FLIGHT/TEST TAPES

The airborne computer is loaded by means of punched paper tapes read in through a computer control console located in the blockhouse. Three tapes are employed: a flight program tape, a constants tape, and a target tape. The flight program is the key element; it embodies the guidance philosophy, control laws, basic computational instructions, and overall executive logic.

The flight program tape instructs the airborne computer how to operate on input and sensed data to meet mission objectives. The constants tape has approximately 400 constants, which are vehicle- and equipment-dependent and are valid for a wide range of trajectories. The target tape consists of 25 mission-dependent parameters. This segmentation of tapes provides operational flexibility. Last-minute minor mission changes can be made by changing only the target tape.

In addition to the three flight tapes, a comprehensive testing routine has been developed that can be used in total vehicle checkouts at the launch base as well as the factory.

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*A more detailed description of Agena software is contained in LMSC-A991366, Agena Tug/OOS Baseline Software and Ground Support Equipment, Lockheed Missiles & Space Co., Inc., Sunnyvale, Calif, 23 Aug 1971.

5.1.1 Flight Program

The flight program embodies a modular concept in that an executive program ties together 61 separate subroutines. This modularization permits easy access to individual elements if mission/vehicle peculiarization is necessary. Enhancing the quick-change provision is the explicit nature of the guidance approach that makes the program applicable to the widest range of missions. The functions of the flight program are illustrated in Fig. 5-1. As shown, the two major segments of programming are the programs related to preflight and the programs related to flight.

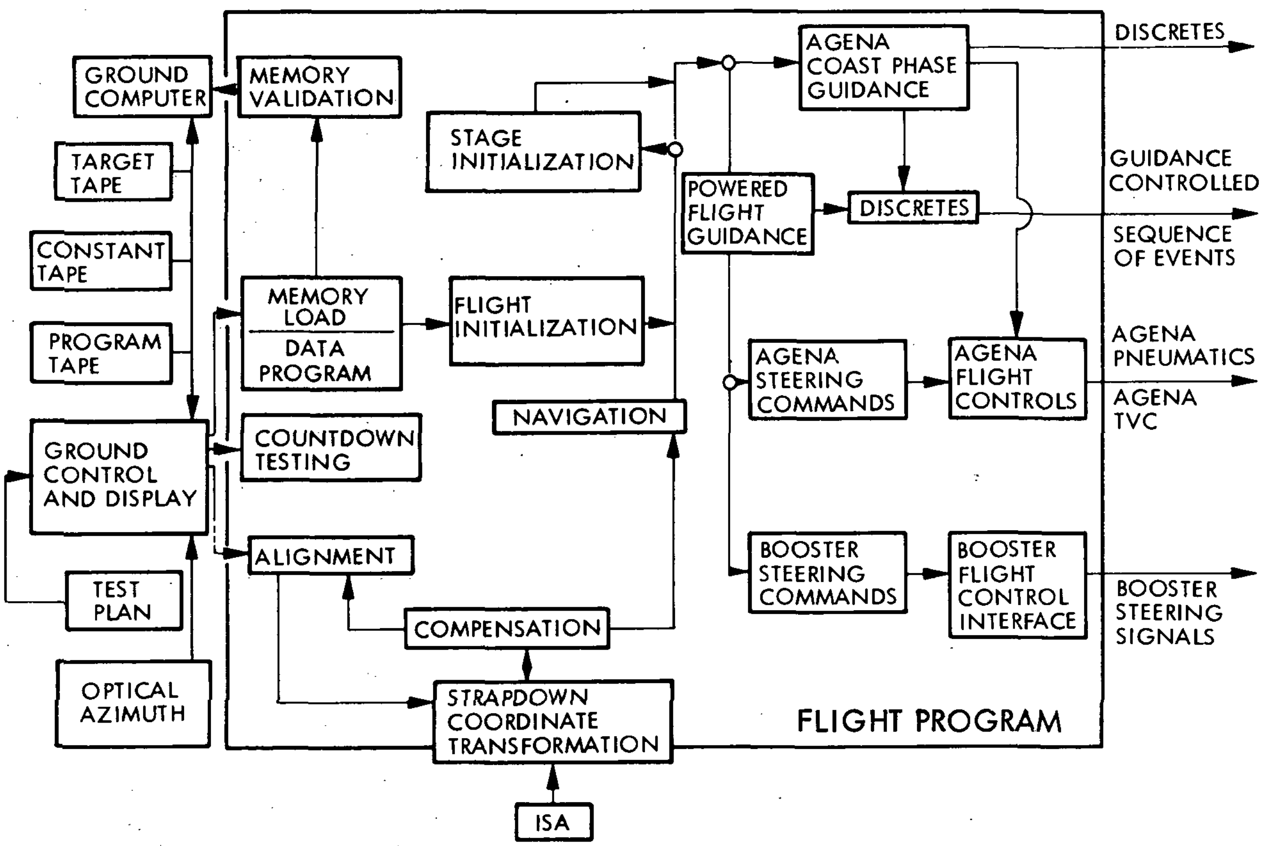


Fig. 5-1 Flight Program Functions

The preflight program segment performs all required status checks and alignments. Memory load, vehicle validation, countdown testing, and, to some extent, flight initialization are vehicle and equipment status checks. Booster steering interfaces are checked, including engine gimballing. The program also self-checks the computer and directs readout of the memory contents over the telemetry link several times prior to launch. A self-contained alignment routine using the earth rotation and the local gravity vector determines vehicle orientation with respect to a guidance reference frame prior to liftoff. The routine is flexible and can be used at any pad or test area by inserting the pad latitude and approximate orientation data. An efficient Kalman filter processes accelerometer and gyro signals, which are contaminated by unknown gyro drift rates, accelerometer bias, and noise in the form of wind-induced random vehicle sway, to give a prelaunch level and azimuth alignment. An external optical azimuth update may be employed to further increase azimuth precision. The inflight segment of operation commences on a command from the ground (Go-Inertial discrete) that causes the program to perform flight initialization.

Measurements of attitude and velocity changes by the inertial sensor assembly (ISA) are transformed into the inertial reference frame 1800 times per second. The transformation is performed automatically by a hard-wired sequencer section in the computer. Standard computer algorithms are used during each flight control cycle to compensate for calibrated values of gyro and accelerometer instrument errors. Algorithms specially developed for strapdown inertial guidance are used in every flight control cycle to compensate for some of the errors induced by the effect of the ascent vibration and acoustic noise.

The navigation routine is exercised once each second, operating on the accumulated compensated values of incremental velocity (ΔV) in the inertial reference frame. The ΔV and a first-order gravity model are used to obtain new position and velocity vectors. This navigation algorithm is commonly used with inertial guidance systems. Scaling permits terminal conditions up to 44,000-nm radius and 65,000-fps velocity.

Stage-initialization routines make possible a common powered flight guidance routine for all launch vehicle stages. Stage-dependent active guidance variables are initialized by the routine at the beginning of each stage control phase. Capability is provided for

up to two booster stages as well as two Agena burns. The algorithm mechanized in the powered flight guidance routine is the commonly used and tested linear sine law. Current position and velocity data from the navigation equations are used each second, and desired time-phased attitude and time to go to cutoff are generated. Booster stages are guided to terminal target radius (R) and radius rate ($\dot{R}$) conditions with shutdown controlled by either fuel depletion or command shutdown. The latter is controlled by energy or angular momentum. The Agena, unlike the booster, is guided to prescribed orbit conditions. Optimum values of R and $\dot{R}$, dependent on the required orbit and the achieved state vector, are determined and used in the linear sine law to generate desired attitude data as a function of time for the steering and flight control routines.

During coast phases, desired inertial attitudes can be achieved and precision attitude maneuvers may be preprogrammed. The Agena flight control routines are digital mechanization of proven attitude control loops. Capability for a variable gain to compensate for moment of inertia and lever arm changes that occur during burn is included, providing excellent phase and gain margin characteristics. Pulse-width-modulated signals are generated by the booster flight control routine for the booster autopilot.

The discrete routine is controlled by the constants on the flight constants tape and may be either time dependent, such as time from Go-Inertial discrete time from stage beginning, or trajectory dependent. Examples of trajectory-dependent discrete control are fairing ejection as a function of altitude and stage engine shutdown at predetermined angular momentum.

Data from the computer that are to be telemetered are formatted, stored, and prepared for transmission by the telemetry routine. Computer calculated quantities (i.e., navigation, steering, control, intermediate guidance calculations, and status words) are examples.

5.1.2 Test Program

Like the flight program, the integrated test tape (ITT) is modular, consisting of over 100 subroutines. Most of the subroutines are commonly used routines for checking computer-controlled subsystems. Several are identical to status check routines in

the flight program. Figure 5-2 illustrates the functions contained in the ITT. As indicated, the ITT addresses the airborne computer through the computer control equipment, which allows manual vehicle/guidance checks as desired.

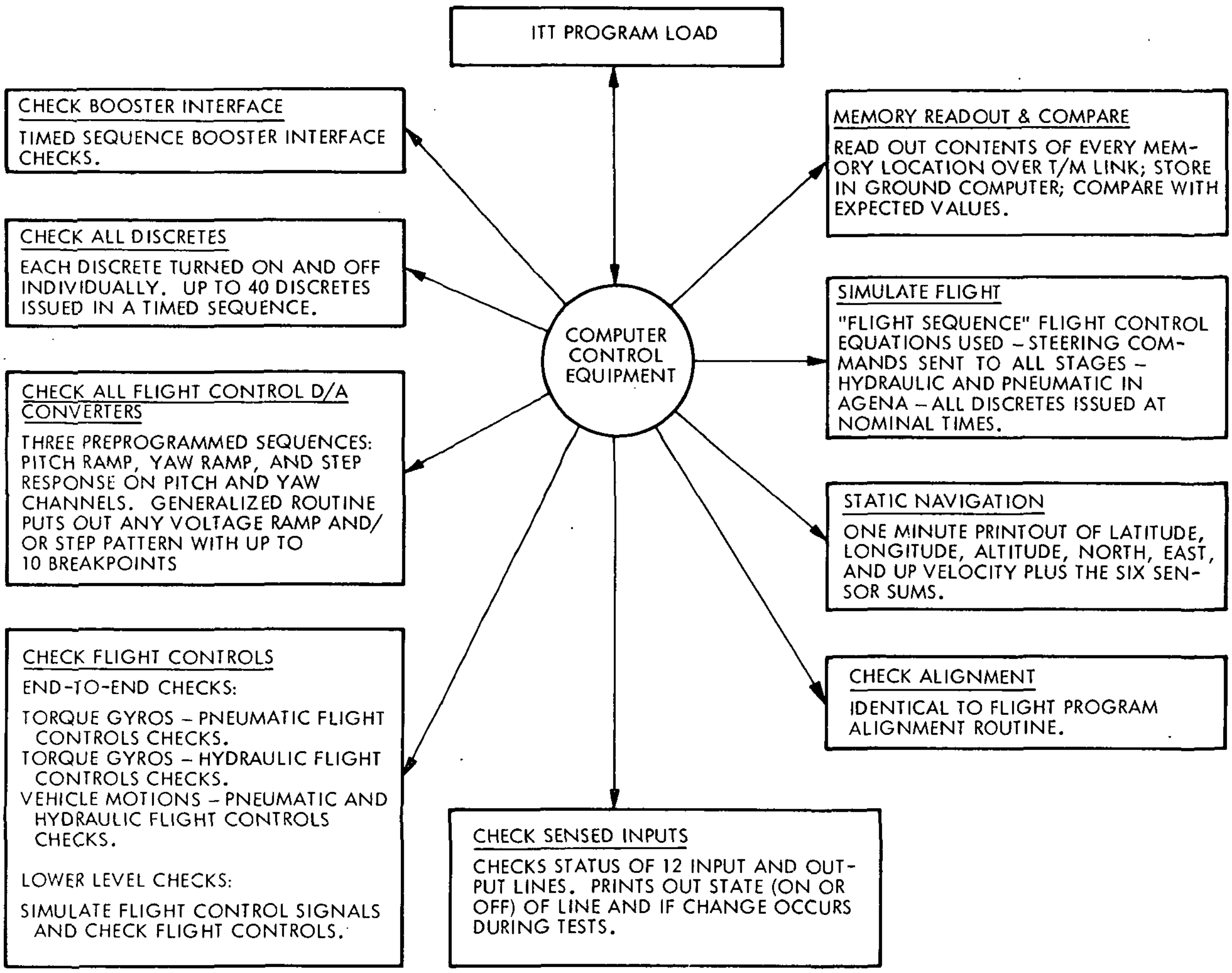


Fig. 5-2 Preflight Checks

5.2 VERIFICATION AND VALIDATION

Successful flight program operations are assured by a straightforward procedure employing advanced software tools. The method of verification and validation and the tools are briefly described in the following paragraphs. A representation of the process is illustrated in Fig. 5-3.

5.2.1 Requirements

The flight program package for Agena is a set of flexible, modularized routines that can satisfy a broad range of mission applications. In most cases, the existing structure of the program can perform the mission with simple changes to constants tapes and target tapes. However, before applicability of existing programs can be validly determined and before mission applicable constants and target parameters can be defined, a set of mission requirements must be developed. Mission requirements include orbit characteristics, launch vehicle configuration, launch range, and such constraints as maximum loads, launch probability, launch windows, etc. These requirements are defined in a Guidance Technical Requirements document. Once developed, a guidance specification is written to document the manner in which these requirements and any mission/vehicle-peculiar requirements are implemented.

5.2.2 Trajectory

The mission-requirements data are used to develop a trajectory that can achieve the mission objectives by a proven, standard trajectory simulation program for the UNIVAC 1108 at LMSC's large digital computer facility. The program simulates ascent motions and works in conjunction with an optimization program to define a trajectory that results in the greatest payload weight on orbit for the given vehicle and mission. The defined trajectory satisfies objectives for intermediate and final flight parameters. The optimization is constrained by vehicle limitations, such as maximum heating, asymmetric heating limits, structural loading limits, maximum turning rates, etc. The developed nominal trajectory is used as a tool to test the flight program. Design trajectories are generated by the simulation program to be

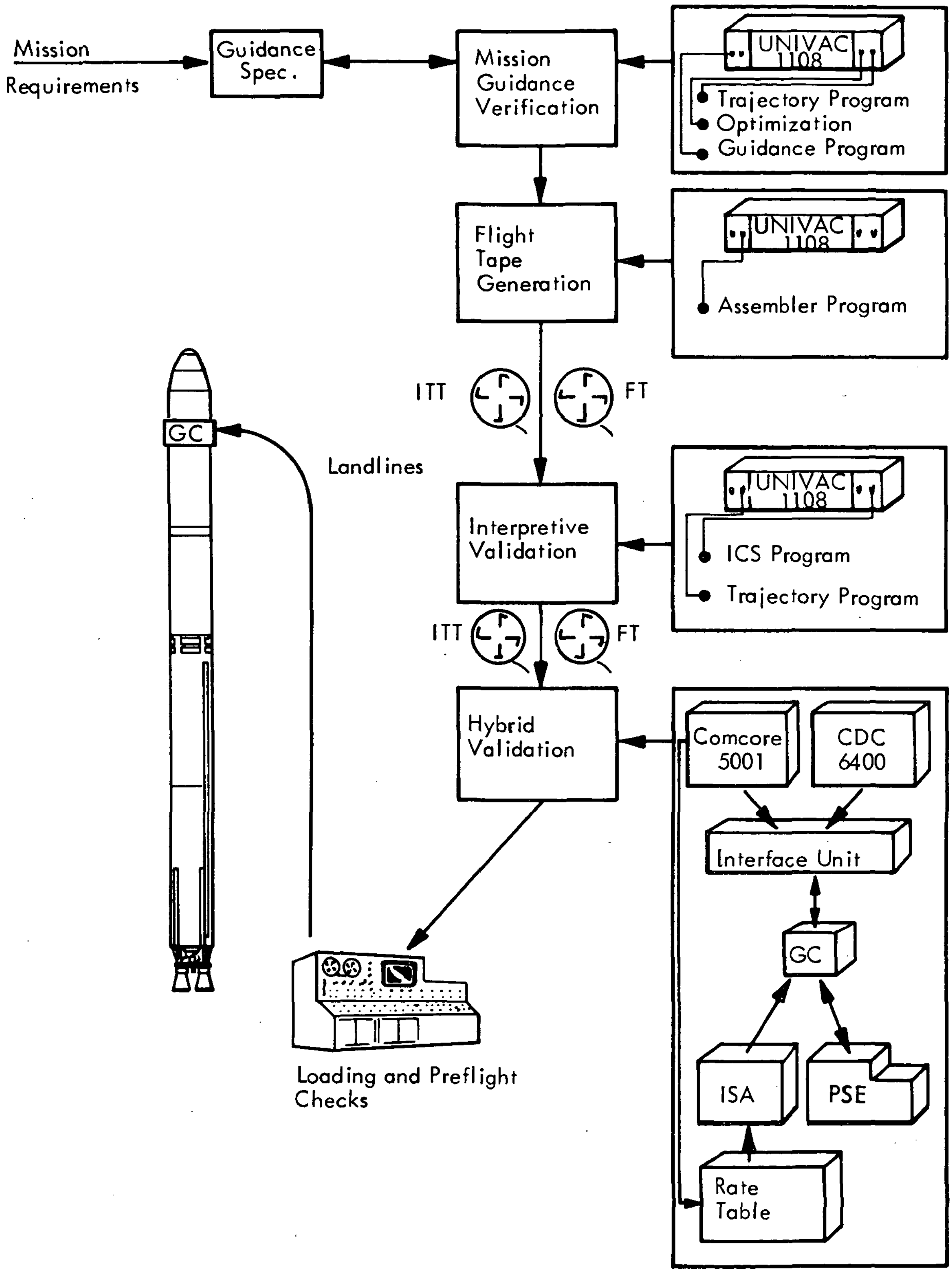


Fig. 5-3 Software Overview

used for analysis of loads, dynamics, and thermodynamics. A nominal reference trajectory can also be provided for use in planning for range and communications coverage.

5.2.3 Guidance Simulation and Verification

Routines coded from the guidance specification have been developed which, when coupled with the trajectory program, permit generation of a guided flight simulation by the UNIVAC 1108. This comprehensive simulation verifies all computation algorithms to support navigation, coordinate system transformation, guidance laws, steering laws, flight controls, discrete formulation, compensation, and general mathematics. The trajectory and guidance simulation can be operated together at all airborne computer cycle frequencies so that each mode of calculation can be checked and compared with the requirements of the mission.

Once the guidance simulation is verified, the equations are combined with the airborne computer peculiar requirements and coded in assembly language for the assembler program that generates the actual binary instructions for the airborne computer. This program also assigns all undesignated storage locations and contains program checking provisions. The instructions for the airborne computer are recorded on a magnetic tape that is converted to the actual flight paper tape on an IBM 360-30 computer.

5.2.4 Interpretive Validation

If the flight program is modified, each modified segment is tested in detail to establish that the modification can be performed without jeopardizing the total program and to establish that the modified segment operates satisfactorily. The tool used for this test is an interpretive computer simulation (ICS). The ICS contains a simulation of each detail of airborne computer operation. Instructions, bit manipulations, processor, and all elements that make the airborne computer perform its functions are duplicated

and subjected to the program instructions used to generate the flight tapes. Each input of information that the airborne computer receives in flight is also simulated. These inputs are the gyro and accelerometer data that result from "flying" the prescribed mission. In effect, the total mission is simulated as it relates to airborne computer operation. The simulation of the airborne computer operation executes at one-sixth normal speed. The ICS permits a thorough evaluation of computer events and a measurement of total simulated computer performance. System timing, scaling, addressing, word length, computation round off, and interrupt processing are among the parameters evaluated. The result of this testing is a validated flight program as coded for the airborne computer.

5.2.5 Hybrid Validations

A final test assures that the flight tapes and the airborne hardware are flight ready. This test is a hybrid validation in which flight hardware is placed on a motion simulator with use of a digital computer and an analog computer team to "fly" the mission. The flight tapes are used to program the airborne computer with the same type of equipment contained in the blockhouse; liftoff is simulated. The resultant flight profile is then evaluated by the digital computer. A successful flight provides the final assurance that the software and the airborne computer are compatible and operate correctly.

Section 6

GROUND SUPPORT EQUIPMENT

Section 6
GROUND SUPPORT EQUIPMENT

The ground support equipment for the Ascent Agena is essentially of the same type (and in most cases is existing equipment) as that used with the SS-01B Standard Agena. GSE is categorized as servicing, handling, and checkout and test equipment, as defined below.

Servicing Equipment: Supports the Agena during on-pad activities, including testing, propellant gas loading, and countdown to launch. It includes:

- Portable Air Conditioners. Supply temperature- and humidity-controlled air for vehicle and payload temperature control.
- Propellant Transfer Units. Temperature-conditions and transfers both fuel (unsymmetrical dimethylhydrazine, UDMH) and oxidizer (inhibited red fuming nitric acid, IRFNA), into the vehicle.
- Pneumatic Control Cabinets. Controls, pressure-regulates, and gages the loading of helium and nitrogen (either locally or remotely) into the Agena gas spheres. It also provides pneumatic control for solenoid-operated valves and switches in the propellant transfer units and the vent valving box; furnishes control signals for launch stand functions such as boom retract, umbilical release, and lanyard release; and provides power for the blowoff of pneumatic quick-disconnect umbilicals.
- Umbilical System Installation. Connects the launch stand GSE units to the vehicle. Major components are the plumbing lines, vent safety relief valving box, umbilical retractors, and electrical cables.

Handling Equipment: Handles and supports the Agena while it is assembled, disassembled, and transported. It includes:

- Vehicle Handling Yokes. Support the vehicle on the equipment used during handling, storage, checkout, and mating operations.

- Slings. Provided for hoisting in a vertical or horizontal position and tilting the vehicle and engine and for removing the yokes in any position.
- Yoke Beam Assemblies. Attached between the forward and aft yokes to reinforce the vehicle tank section when hoisting forces are applied to the vehicle handling yokes.
- Primary Mobile Units of Equipment. The vehicle handling dolly is a movable workstand (also used for in-plant transportation) that supports the vehicle in a horizontal position. The vehicle transporter and auxiliary equipment (tank pressurization, IRP heater, and nylon cover) are used with trucks and in cargo aircraft to anchor, support, and shock-mount the vehicle during vehicle shipment.

Checkout and Test Equipment: Used to perform vehicle subsystem tests, vehicle validation tests, and launch base tests, including countdown. These tests make use of temperature/altitude chambers, acoustic test cells, vibration equipment, automatic test complexes, and data processing facilities.

6.1 GROUND SERVICING EQUIPMENT

Ground servicing equipment is composed of the following equipment integrated into a system to support vehicle servicing and launch:

a. Two primary propellant transfer units

b. One pneumatic system (regulates and distributes N_2, He, and guidance gas to the vehicle)

c. Vehicle and payload air-conditioning system

d. Pneumatic service lines and equipment

e. Vehicle and facility leak detectors

f. Umbilical system

Table 6-1 is a summary list of servicing equipment for the Ascent Agena that is used in the system test area at the factory and at the launch complex. The "installation"

items are peculiar to the specific launch pad configuration at SLC-4W and include the umbilical lines up the mast.

Table 6-1

SERVICING EQUIPMENT FOR THE ASCENT AGENA

Nomenclature	Dwg No. (or Similar to)	System Test	Launch Complex	Total Qty
Piping Installation (Fuel Room)	1591224		X	1
Piping Installation (Oxidizer Room)	1591225		X	1
Detector Installation	1597012		X	1
Umbilical System Installation	1562999		X	1
Umbilical Boom Assembly	1563784		X	1
Propellant Chiller Assembly	1511299		X	1
Fuel Transfer Unit	1554058		X	1
Oxidizer Transfer Unit	1554058		X	1
Gas Pressurization Equipment	1592891		X	1
Gas Pressurization Cabinet	1593203		X	1
Pneumatic Control Cabinet	1593207		X	1
Service Lines & Equipment Installation	1563652		X	1
Air-Conditioner Installation	1562605		X	1
Hydraulic Test Cart	1062637	X		
Gas Pressurization Cabinet (Freon)	1518321		X	1
Vent Safety Box	1595482		X	1
Console (Gas Pressurization Checkout)	1591790	X		1
Hydraulic Pressure Unit	1553002	X		1
Console (High-Pressure Gas Control)	1551281	X		1
Gas Compressor Assembly	1521941	X		1
Hydraulic Test Cart	1460383	X		1
Control Cabinet Pneumatic Test	1500973	X		1
Cart Assembly (Propellant Loading)			X	1
Chiller Assembly (Battery Cooling)		X	X	2
Optical Azimuth Orientation Alignment System Installation	1563648		X	1

6.2 GROUND HANDLING EQUIPMENT

The ground handling system provides equipment to assemble, disassemble, transport, transfer, load, unload, store, lift, and protect the Ascent Agena. Handling equipment for the Ascent Agena in the factory and at the launch complex is listed in Table 6-2. Should it be desirable to transport the Agena with payload installed, an existing transporter (Drawing 8100168) used on another program, may be applicable. This transporter provides environmental control and can accommodate loads up to 10 ft in diameter, 70 ft long, and weighing up to 30,000 lb.

Table 6-2
HANDLING EQUIPMENT FOR THE ASCENT AGENA

Nomenclature	Dwg No. (or Similar to)	Assembly Area	Test Area	System Test	Acoustic Chamber	Pre-Ship Area	In Transport	Launch Complex	Total Qty
Dolly (Vehicle Handling)	1503447	X		X		X		X	4
Dolly (Vehicle Storage)	1585127	X		X		X			1
Tilt Sling (Vehicle)	1503703	X	X	X	X	X		X	2
Sling (Yoke Removal)	1506526	X						X	2
Transporter (Vehicle)	1503448						X		1
Towbar (Handling Dolly)	1506532	X		X		X		X	4
Sling (Horizontal Hoisting)	1512162	X		X		X		X	2
Crane Control, 5 Ton	1513234							X	1
Yoke Beam (Vehicle Handling)	1518970	X		X			X	X	6
Fixture (Vertical Assembly)	1506501	X							1
Vertical Workstand	1503450	X							1
Dolly (Guidance Module Handling)	1543906			X					1

Table 6-2 (Cont)

Nomenclature	Dwg No. (or Similar to)	Assembly Area	Test Area	System Test	Acoustic Chamber	Pre-Ship Area	In Transport	Launch Complex	Total Qty
Stand (Vehicle Checkout)	1508697	X							1
Dolly (Engine Removal)	1556389	X							1
Stand (Vehicle Pitch & Roll)	1503443			X					1
Fixture (Battery Installation)				X				X	2

6.3 CHECKOUT AND TEST EQUIPMENT

A factory-to-launch concept is used for Ascent Agena checkout and test equipment design. The factory vehicle system tests verify system-level requirements necessary to assure a launch-pad-ready vehicle. Only those servicing and installation operations not feasibly performed at the factory are accomplished at the launch base. Figure 6-1 depicts a typical baseline Ascent Agena launch stand test arrangement.

6.3.1 Electrical GSE

The electrical GSE used at the launch complex for the Ascent Agena is listed in Table 6-3.

6.3.2 AGS Test Equipment

The following ground test equipment is used for IGS/AGS tests and calibrations in the vehicle system test (VST) and launch complex test areas:

- System Test Set (STS)
- Automatic Data Set (ADS)
- Interface Test Unit, GC (ITU)
- Interface Test Unit, ISA (ITU)
- Cable Set
- Special Cable Set
- ISA Holding Fixture

Brief descriptions follow. Refer to Fig. 6-2 for the interconnect diagram.

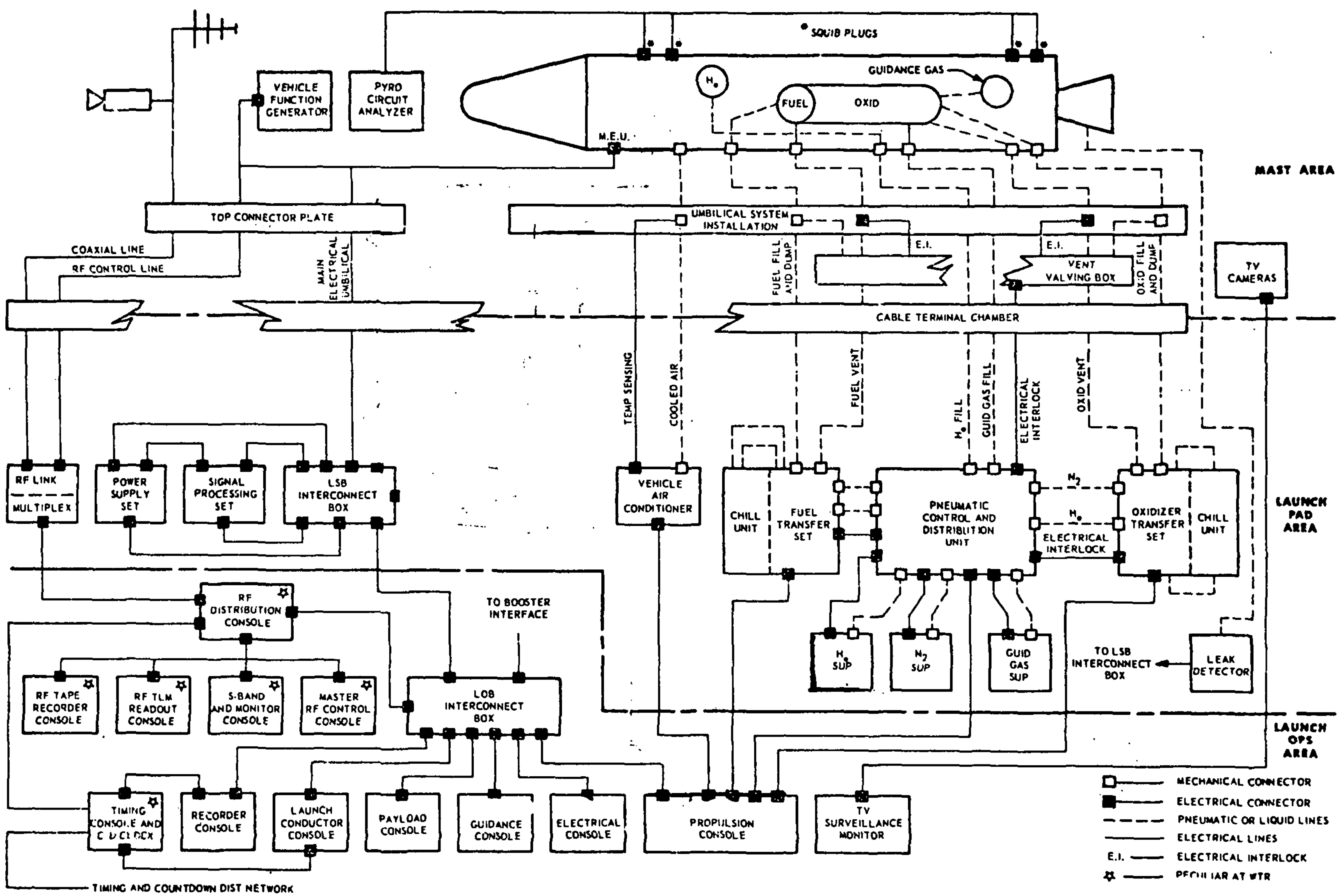


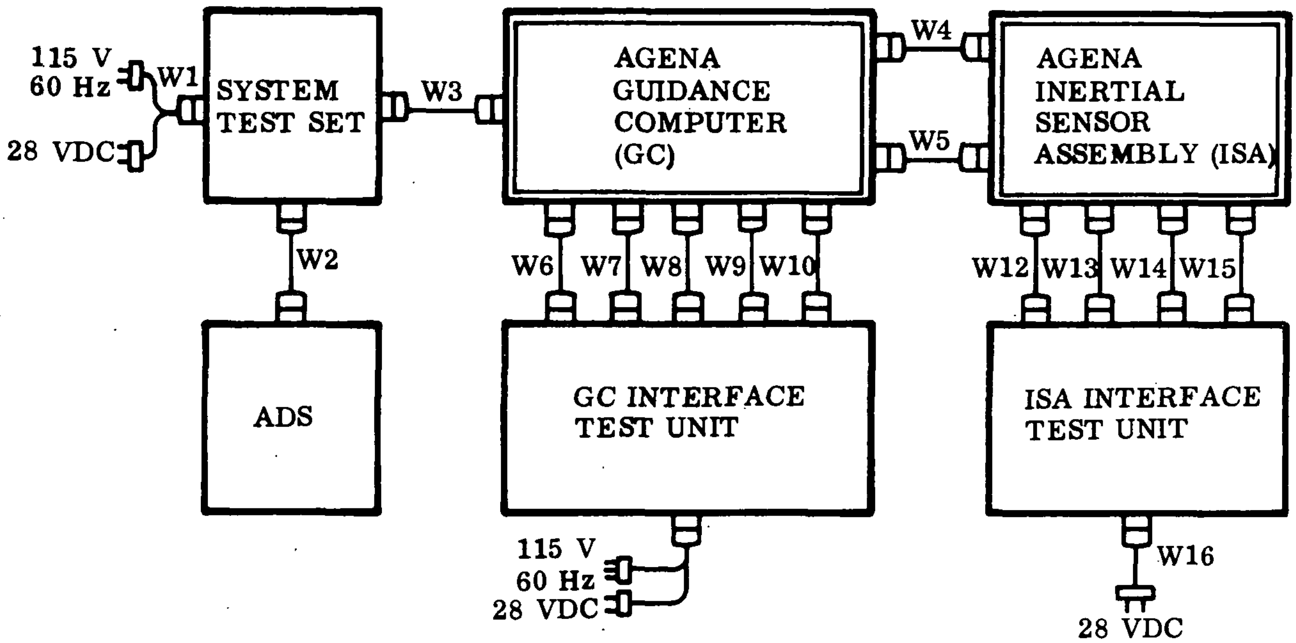
Fig. 6-1 Typical Baseline Ascent Agena Launch Stand Test Arrangement

Table 6-3

LAUNCH COMPLEX ELECTRICAL GSE FOR THE ASCENT AGENA

Nomenclature	Drawing Number (or similar to)
LAUNCH COMPLEX EDL	1563661-501
SGLS Console (LOB)*	1561037-501
Signal-Conditioning Console (LOB)	1562019-501
RFI and Monitor Console (LOB)	1562429-503
Tape Recorder I/F Console (LOB)	1563774-501
Ampex Magnetic Tape Recorder (LOB)	FR 1900
RF Power Distribution Console (LOB)	1562396-501
Supplemental Telemetry Processing Console (LOB)	1556872-505
RF Telemetry Monitor Console (LOB)	1534998-501
PCM Decom/Comp Console (LOB)	155010-503
AGS Console (LOB)	1563676-501
Signal Data Recorder Console (LOB)	1593456-501
Pneumatic and Propulsion Console (LOB)	1526721-515
Elec and Guidance Console (LOB)	1526947-515
Aux Elec Console (LOB)	1561455-501
Launch Cond Console (LOB)	1562730-501
LOB Power Supply Console (LOB)	1593459-505
Timing Console (LOB)	1532023-507
Test Plug Console (LSB)**	1562653-501
Signal Data Processing Console (LSB)	1562695-501
LSB Power Distribution Console (LSB)	1555803-503
Auto Data Set (LOB)	Dwg-8399H1
LSB Power Supply Console (LSB)	1563777-501
LOB I/C Console (LOB)	1562966-501
LSB I/C Console (LSB)	1556604-501
Program Recorder Console (Gantry)	1552319-505
VFG Console (Gantry)	1562722-501
Air Conditioning Control Console (LOB)	1562960-501

*LOB – Launch Operations Building
**LSB – Launch Support Building



		LENGTH (FT)			LENGTH (FT)
W1	STS POWER	20 ± 1/2	W9	GC TM/FCE/HOR	10 ± 1/2
W2	STS/ADS	10 ± 1/2	W10	GC TEST	10 ± 1/2
W3	STS/GC	20 ± 1/2	W11	GC ITU POWER	20 ± 1/2
W4	GC TO ISA	20 ± 1/2	W12	ISA POWER	10 ± 1/2
W5	ISA TO GC	20 ± 1/2	W13	ISA TM	10 ± 1/2
W6	GC POWER	10 ± 1/2	W14	ISA TEST	10 ± 1/2
W7	PULSE DISC	10 ± 1/2	W15	ISA FAST HEAT	10 ± 1/2
W8	VAR DIS	10 ± 1/2	W16	ISA ITU POWER	20 ± 1/2

Fig. 6-2 System Interconnection Block Diagram

System Test Set. The system test set is contained in a single wire rack and consists of a system control unit, power control unit, and a power supply. The test set uses 115 volts, 60 Hz, at 5 amp and 28 vdc, 300 watts.

Automatic Data Set (ADS). The Kleinschmidt automatic data set (Model 321) provides the functions of typewriting and tape reading and punching for the vehicle computer initialization. It is a table-mounted, duplex send-receive printer with punch and reader attachment. The ADS is used in conjunction with the system test set to load and verify loading of the vehicle computer memory, punch tapes, and record data on punched tapes or printed paper. Power requirements are 115 volts, 60 Hz, at 250 watts.

Interface Test Unit (GC). The interface test unit is a table-mounted panel that contains the circuitry control switching to simulate the interfaces of the vehicle computer. It will permit complete checkout of the computer under simulated operating conditions prior to subsystem tests. The test set simulates the flight control system, ISA, telemetry discretes, and power supply. It operates on 115 volts, 60 Hz, at 0.25 amp and 28 vdc at 12 amp.

Interface Test Unit (ISA). This unit is similar in function to the guidance computer unit and operates from 28 vdc at 8 amp. It is typically a flat-panel, table-mounted cabinet.

Cable Set. The cable set consists of four interconnect cables that are required to connect the AGS to the ground test equipment. These interconnects are between the Kleinschmidt automatic data set and the system test set (10 ft) and between the system test set and the AGS computer (20 ft).

Special Cables. Primary power cables from the facility 28 vdc and 115 volt, 60 Hz, outlet boxes are fabricated. Additional cables are needed to interface the PSE equipment and the vehicle system test equipment.

ISA Holding Fixture. This unit is a two-degree-of-freedom gimballing assembly designed to mount the ISA assembly during performance of six-position inertial component tests. It is used in conjunction with, and mounted upon, a precision rate table.

The holding fixture provides the capability of mounting the ISA in any position with respect to the earth's rotational axis, the rate table rotational axis, or the earth gravity vector.

The STS and ADS are used together to provide ground control of the IGS during Agena system tests and launch base operations, as well as computer-level tests. The interface test units and cable set are used for testing the individual GC and ISA. The holding fixture provides capability to position the ISA on a positioning or rate table as required during component testing.

The STS consists of a system control unit (SCU) and a power control unit (PCU), each in its own slide-mounted chassis. The PCU is essentially a power switching unit with overload circuit breakers that control external 115-volt, 60-Hz power and 28-vdc power to the SCU. The SCU, which provides the operational interface with the IGS, contains a power supply providing +5 vdc (logic power), ±12 vdc (interface power), and +10 vdc (lamp power).

Ground operations at the vehicle system level using only the STS and ADS are discussed further. The SCU is electrically connected to the GC and ISA through landlines, which are about 2300 feet long at the launch site, through the Agena vehicle electrical umbilical.

6.3.2.1 System Control Unit (SCU) Description. The SCU has a front panel for control and display, arranged in groups of functionally related items (Fig. 6-3).

The top row of lights is a display of the binary word sent from the GC. The next row is a display of the binary word sent to the GC. A bank of toggle switches is under the second row of indicator lights. Associated with each bit of the display is a toggle switch used to set the bit in the word to be sent.

In the next row down, on the left is an ADS group divided into three subgroups: Compare (ERROR/RESET), Load (CTS/VERIFY/LOAD), and Reader (ADVANCE/AUTO/MANUAL). The ADS group is discussed later.

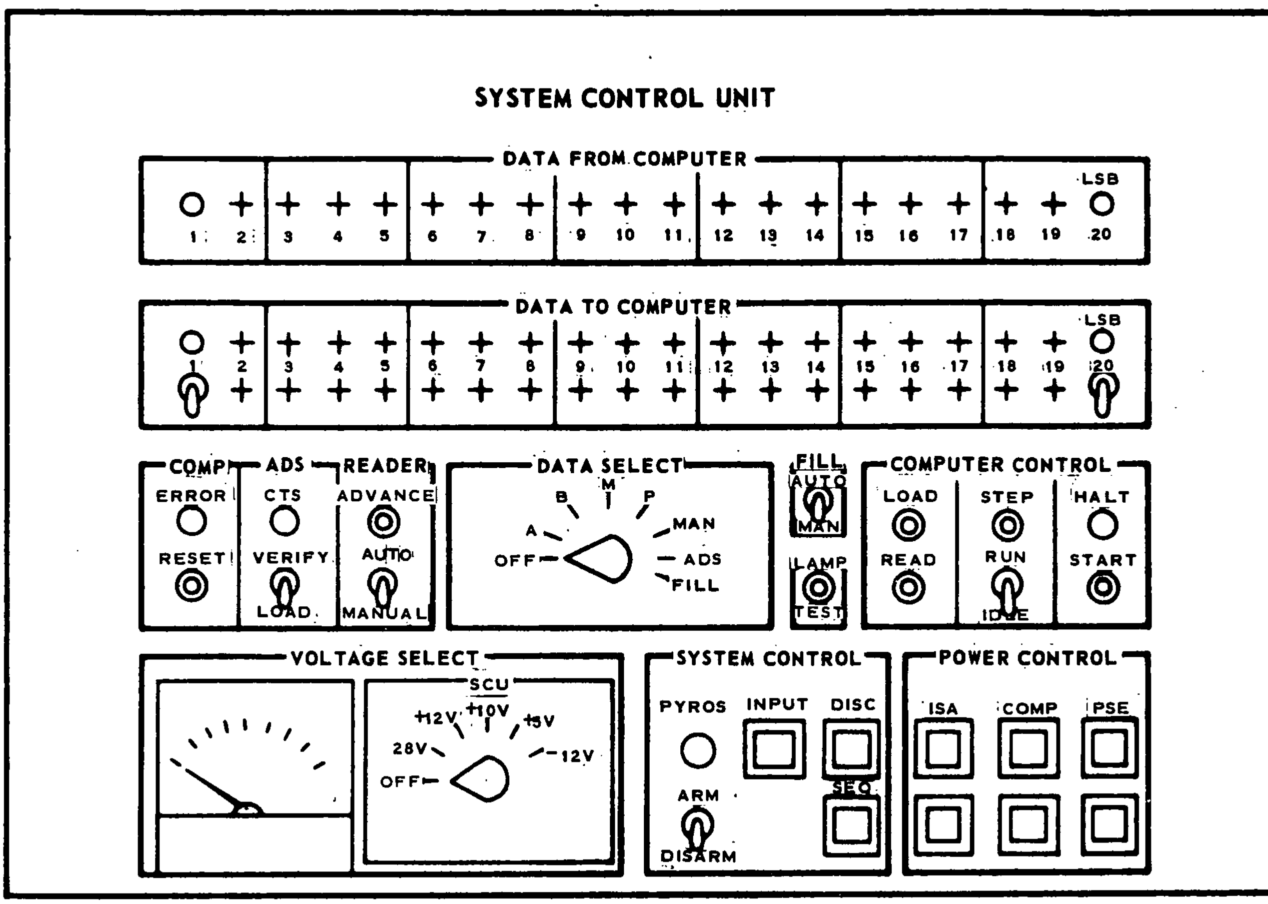


Fig. 6-3 System Control Unit Front Panel

The Data Select group consists of a switch used to select one of eight positions: OFF (no selection), A (A register in GC), B (B register in GC), M (Master register in GC), P (Program register in GC), MAN (Manual data loading mode) ADS (Operations with ADS enabled), and FILL (fill function enabled).

The Fill group has a toggle switch that selects AUTO (Automatic mode) or MAN (Manual mode) of fill operation. A lamp test button to test the panel lamps is located below the Fill group.

The Computer Control group consists of three subgroups: LOAD and READ pushbuttons, STEP pushbutton and RUN/IDLE toggle selector switch, and HALT indicator and START pushbutton.

The Voltage Select group on the bottom row has a voltmeter and an SCU voltage selector switch for SCU voltage checks.

The System Control group consists of four subgroups: PYROS (Pyro ARM indicator and Pyro ARM/DISARM selector toggle switch), PSE Input (Enable/inhibit PSE inputs to GC alternate action split screen switch), ISA Power DISC (power discrete inhibit/enable alternate action split screen switch), and SEQ (Sequencer inhibit/enable alternate action split screen switch).

The Power Control Group consists of three subgroups: ISA (ISA On indicator switch, and ISA Off indicator switch), COMP (GC On indicator switch, and GC Off indicator switch), PSE (PSE On indicator switch and PSE Off indicator switch).

A block diagram of the SCU is shown in Fig. 6-4. The manual control, toggle switches, data "From Computer" display, and data "To Computer" display boxes and the Halt, Compare Error, and Arm Pyros lamps represent items on the front panel. Interfaces of the SCU with the I/O of the GC, the power supply of the GC, and the ADS are shown with an identifier associated with each line function.

There are two serial registers with bit-for-bit parallel connections with associated front panel display lamps. Gating circuits are associated with the Data to Computer toggle switches that set up the manual data input. A Timing/Control element provides a 5-kpps clock pulse rate, timing generator, and control logic. A six-bit comparator is set up to energize the Error lamp if a mismatch between compared bits is present. A Fill program permanent circuit is provided.

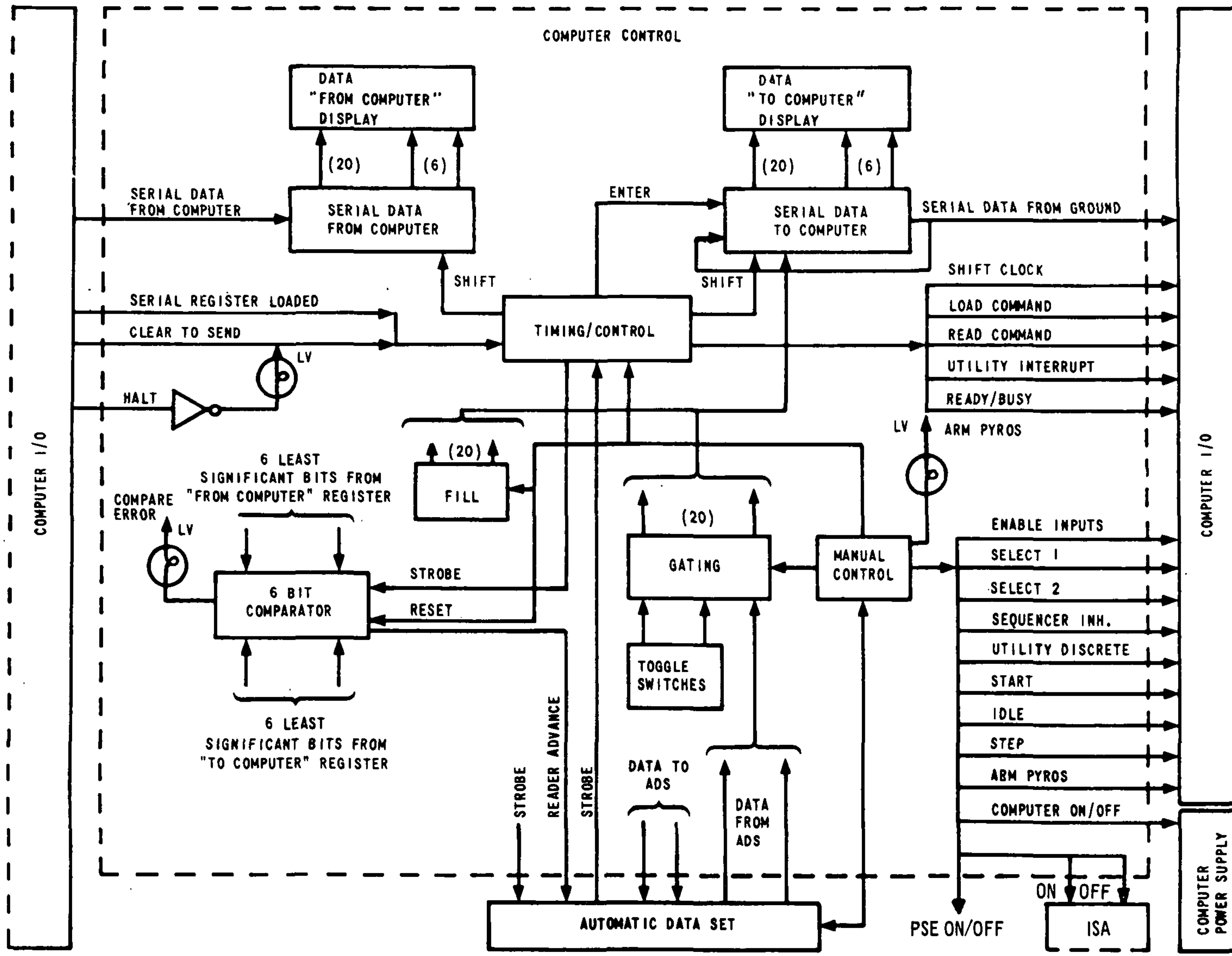


Fig. 6-4 System Control Unit Block Diagram

6.3.2.2 SCU Operation. The following operations can be performed with the SCU:

a. Turn PSE power on and off
b. Turn GC on and off
c. Turn ISA on and off
d. Inhibit and enable inputs to GC from SCU
e. Inhibit and enable GC sequencer
f. Arm and Disarm pyro pulse discrete outputs from I/O
g. Put GC into Idle mode
h. Advance GC one instruction at a time
i. Put GC in Run mode
j. Load 20-bit words into A, B, or M register
k. Read contents of A, B, M, or P register
l. Load 20-bit word in GC memory
m. Reset GC to an initial condition (START)
n. Fill a bootstrap data loading program into the GC
o. Test SCU lamps

Power control interlocks interrelating items a., b., and c. above are as shown in Fig. 6-5. Note that the ISA ON indicator source is 800 Hz from the ISA. The GC is turned on when the switch is open. The same result is achieved by pulling the electrical umbilical.

The Enable Inputs control signal (item d.) actuated by depressing the PSE INPUT switch in the System Control panel group, is sent to the I/O where it is gated with the following other inputs from the SCU to the I/O: Load Command, Step, Idle, Utility Interrupt, Sequence Inhibit, and Start. When the Enable Inputs line is carrying the inhibit signal, the other named inputs are inhibited. The same result is achieved by pulling the electrical umbilical.

The Sequencer Inhibit Control signal (item e.) actuated by depressing the SEQ switch in the System Control panel group, is sent to the I/O where it is gated with the C1 clock input from the ISA. When the Sequencer Inhibit line is carrying the inhibit signal, the C1 clock pulses are inhibited from causing the Sequencer Interrupt to occur. If the switch is in the ENABLE position, or if the electrical umbilical is pulled, the Sequencer is enabled.

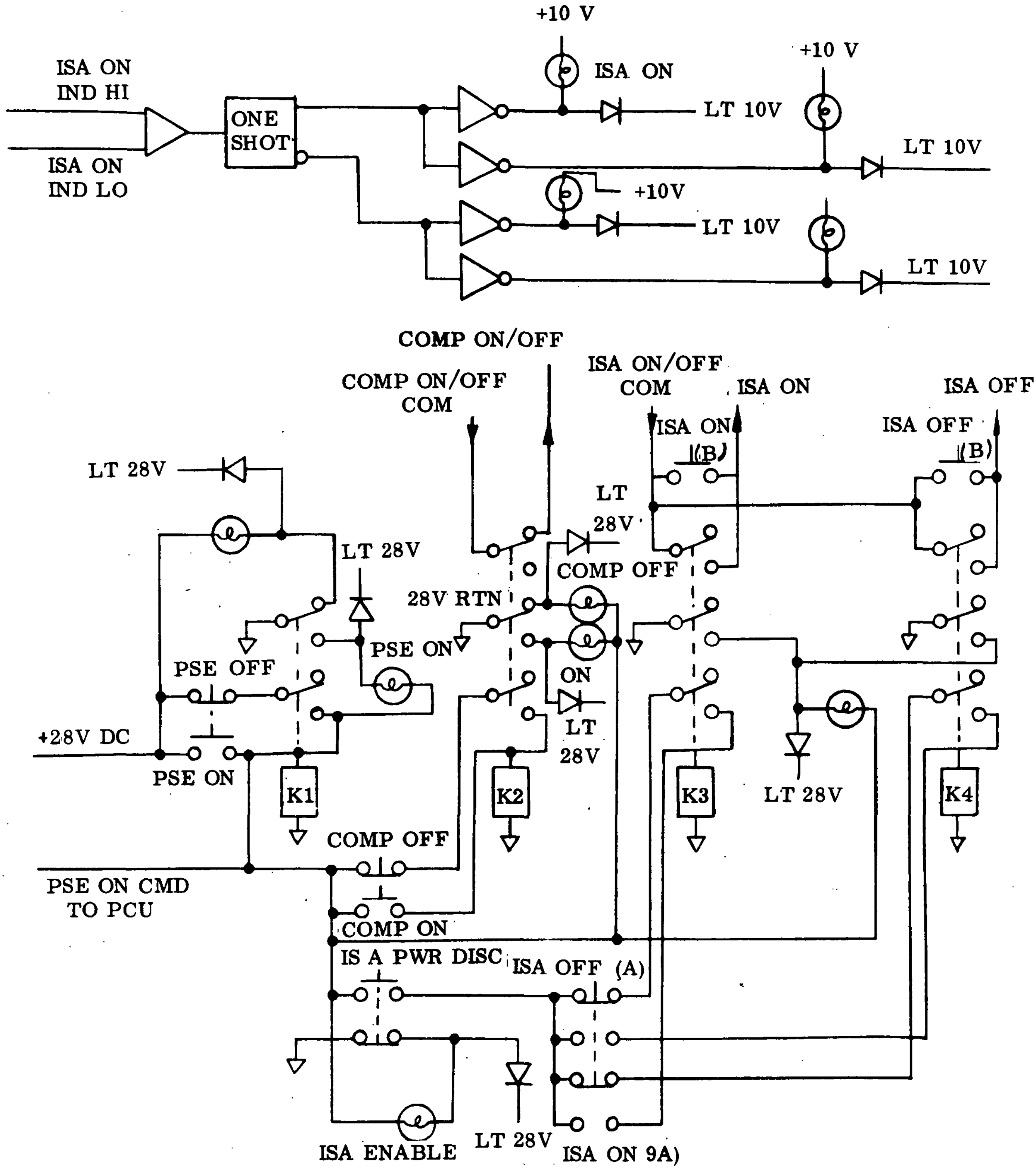


Fig. 6-5 System Control Unit Power Control Interlocks

The Arm Pyros control signal (item f.) actuated by throwing the pyros ARM/DISARM toggle switch in the System Control panel group, is sent to the I/O where it is gated with the 21 pyro pulse discrete outputs from the GC. When the Arm Pyros line is carrying the disarm signal, pulse discretes will not be issued from the GC even if they are generated internally by the program. If the switch is in the ARM position, or if the umbilical is pulled, the pyros are armed.

The Idle control signal (item g.) actuated by throwing the RUN/IDLE toggle switch to IDLE in the Computer Control panel group, is sent to the I/O where it holds the GC processor in the Idle mode. In this mode, the processor is in the HALT state.

The Step control signal (item h.) is actuated by depressing the STEP pushbutton in the Computer Control panel group only when the RUN/IDLE switch is in the IDLE position and is locked out when the RUN/IDLE switch is in the RUN position. The Step signal is sent to the I/O where it causes the next instruction to be executed. A Read cycle is initiated 100 microseconds after the Step command.

When the RUN/IDLE toggle switch is in the RUN position (item i.) neither an IDLE nor a STEP signal is sent to the I/O where a Run/Idle/Step logic circuit puts the GC in the Run mode. The same result is achieved by pulling the umbilical.

To load the A, B, or M register (item j.) the following procedure is used. The GC is put into the Idle mode. The DATA SELECT switch is set to the selected register, which sets up a logical code on the Select 1 and Select 2 lines to the I/O as follows:

If Selected Register is	Select 1 is	Select 2 is
A	0	0
B	1	0
M	0	1
P	1	1

A logic circuit selects the register from the conditions in the table. The 20-bit data word to be sent to the selected register is then manually set up on the bank of toggle switches.

Next, the Load cycle is initiated by depressing the LOAD pushbutton in the Computer Control group. This enables the 5-kpps shift clock input to the timing generator. The Ready/Busy line goes Busy and remains Busy until the end of the load process.* When the first clock pulse to the timing generator falls, an Enter signal is developed in the control logic which sets the Data to Computer register according to the setting of the bank of toggle switches. Twenty shift clock pulses are then sent to the I/O serial register and to the Data to Computer register, which transfer the data serially, one bit per pulse, to the I/O serial register and also circulate the data back into the Data to Computer register. A LOAD command is then sent to the I/O, which strobes the data out of the serial register into the register selected by the DATA SELECT switch. When the Load cycle stops, the data are displayed on the SCU Data to Computer display and may be compared to the toggle switch settings. In addition, a Read cycle is initiated, which results in a Data from Computer display of the contents of the loaded register.

To read the contents of the A, B, M, or P register on the SCU Data from Computer display (item k.) the following procedure is used. The register to be read is selected on the DATA SELECT switch. The Read cycle is then initiated by depressing the READ pushbutton in the Computer Control group, which enables the 5-kpps shift clock input to the timing generator. When the first clock pulse to the timing generator falls, a READ command is developed and remains ON (true) until the fall of the second clock pulse. The READ command is transmitted to the I/O and is used there to transfer the data from the selected register in the GC into the I/O serial register. From the I/O serial register, the data are clocked into the Data from Computer register in the SCU. Finally, a strobe signal from the SCU control logic transfers the data from the register to the Data from Computer display.

To load 20-bit words from the Data to Computer toggle switches to the GC in the run mode (item 1.), the following procedure is used. The RUN/IDLE switch is set to RUN. The SCU DATA SELECT switch is set to the MAN (Manual) position and the 20-bit data word is set up on the bank of toggle switches. The LOAD pushbutton in the Computer Control group is then depressed to initiate the Load cycle. The 5-kpps shift clock

*This happens every time the LOAD pushbutton is depressed to start a load cycle or when a strobe is received by the SCU from the ADS.

input to the timing generator is enabled. When the first clock pulse to the timing generator falls, an enter signal is developed in the control logic which sets the Data to Computer register in accordance with the setting of the bank of toggle switches. Twenty shift clock pulses are then sent to the I/O serial register and to the Data to Computer register, which transfer the data serially to the I/O serial register and also circulates the data back into the Data to Computer register. A UTILITY INTERRUPT and a LOAD Command are then simultaneously sent from the SCU, where a Load character interrupt is generated in the I/O. No Read cycle is initiated in this mode. When the Load cycle stops, the data are displayed on the SCU Data to Computer display.

Depressing the START pushbutton in the Computer Control panel group (item m.) causes a Start signal to be sent to the I/O. In the I/O, appropriate flip-flops are reset. In the processor, the Halt (H) state is entered. The Start signal initializes the GC.

The Fill function provides an initial computer load of a group of instructions of 64 words that programs acceptance of tape data from the ADS. The Fill program is hardwired in the SCU for transmission to the I/O, from where it is loaded into memory. The load transmission can be made automatically or step by step. The operation is initiated by placing the RUN/IDLE switch in RUN position, the DATA SELECT switch in the FILL position, the AUTO/MANUAL switch in the Fill panel group in either position, and depressing the LOAD pushbutton. A sequence of alternate Load and Read operations occurs until the 64 words of instructions are entered into the GC. The GC is then ready to accept data from the ADS tape.

6.3.2.3 Automatic Data Set (ADS) Description. The ADS is a table-mounted typewriter, tape reader, and tape punch assembly designated Kleinschmidt Model 321. The basic model is modified to permit remote reader advance and to accept a teletype tape winder. The unit is powered by 115-volt, 60-Hz power.

The operating speed is 27 characters per second, where each character is 6 bits. A character is generated by depressing a typewriter key or reading a row of punched holes in six-hole tape. The printer, keyboard, punch, and reader can be switched individually in any combination between local circuits and online circuits as required.

Online operation in this application means operation into the SCU. ADS control switches and monitors on the SCU front panel are used during operations with the ADS.

Data from the ADS are in code form as 6-bit characters, so a translation to GC 20-bit word format is required somewhere for system compatibility. The translation is performed in the GC in a program called the Basic Operating System (BOPS).

6.3.2.4 ADS Operation. The following operations can be performed with the ADS:

a. Load a 6-bit character into the GC memory from the ADS keyboard

b. Load 6-bit characters into the GC memory under manual control from six-hole tape

c. Load 6-bit characters into the GC memory automatically from six-hole tape

d. Verify the characters loaded into the GC memory

e. Print out data received from the GC

f. Punch tape data received from the GC

For all operations, the GC is in the Run mode; that is, the RUN/IDLE toggle switch in the SCU Computer Control panel group is in the RUN position, and the SCU DATA SELECT switch is in the ADS position.

To load a character into the GC (item a. above), the ADS keyboard is switched online. When the I/O serial register is clear, a Clear-to-Send (CTS) signal will be transmitted to the SCU and displayed on the CTS indicator in the SCU ADS panel group. The CTS signal is also a loading inhibit/enable signal. When a typewriter key is depressed, a strobe is developed in the ADS, which transfers the selected 6-bit character to the 6 least significant bits (LSBs) of the SCU Data to Computer serial register and initiates a Load cycle. A 20-bit word is shifted to the I/O serial register of which the 14 most significant bits (MSBs) are zero and the 6 least significant bits are the character bits from the ADS. In Load cycle, the data are circulated in the SCU Data to Computer

serial register so that the Data to Computer display will indicate the data sent to the I/O. After the data have been shifted into the I/O serial register, a LOAD command is sent to the I/O from the SCU. The LOAD command initiates the Load Interrupt within the I/O, which transfers the data from the I/O serial register to the GC memory.

To load 6-bit characters into the GC in manual mode from punched tape (item h. above), the following procedure is followed. The Fill program is loaded into the GC by either the automatic or manual method. The tape is inserted in position in the Reader. The LOAD/VERIFY toggle switch in the SCU ADS panel group is placed in the LOAD position and the AUTO/MANUAL toggle switch is placed in the MANUAL position. The tape reader in the ADS is switched online. A CTS discrete must be present if the I/O is to accept the Load. Each time the READER ADVANCE pushbutton on the SCU panel is depressed, the Reader will advance one character and stop. The character will be sent to the I/O and then to memory, and the 20-bit word carrying the character code in the 6 LSBs will be displayed on the SCU Data to Computer display by the previously described procedure.

To load 6-bit characters into the GC in the automatic mode from punched tape (item c. above), the procedure is the same as for manual mode loading except that the AUTO/MANUAL toggle switch on the SCU panel is placed in the AUTO position. The Tape Reader will run until the end of the tape is reached or the CTS discrete is removed.

To verify that the GC load from the ADS is correct (item d. above), the ADS operations described are followed except that the SCU VERIFY/LOAD toggle switch in the ADS panel group is placed in the VERIFY position. In the Verify mode, the loaded character is echoed by the program and compared to the character circulated into the Data to Computer serial register in the SCU before issuing a Reader Advance pulse to the ADS. The steps are:

1. An ADS strobe initiates a Load cycle, the character is shifted to the computer, and the character is circulated back into the DATA to Computer Register.

2. The character is entered into memory and is echoed back to the I/O serial register, which generates a Serial Register Loaded (SRL) signal to the SCU.
3. The SRL signal initiates a Read cycle and the data are shifted into the Data from Computer serial register and compared to the Data to Computer serial register in the 6-bit comparator.
4. If the comparison agrees, a Tape Reader Advance signal is issued to the ADS. If the comparison does not agree, the Compare Error lamp on the SCU panel energizes and no Tape Reader Advance signal is issued. A RESET pushbutton on the SCU panel resets the Compare error circuit when depressed, which sends a Tape Reader Advance signal and de-energizes the Compare Error lamp.

To print out data on the ADS sent from the GC, the Printer in the ADS is switched on-line. Then, every time an SRL signal is received by the SCU from the I/O, a Read cycle is initiated and the Ready/Busy sense signal from the SCU to the I/O goes busy. The data are shifted from the I/O serial register to the SCU Data from Computer serial register. The data are then strobed from the SCU to the ADS and a delayed reset is sent to the SCU Ready/Busy circuit. The delay is 37 msec and is selected to allow time for the ADS to perform the print function before receiving the next data input.

6.4 LAUNCH CONTROL AND CHECKOUT EQUIPMENT

The launch control and checkout equipment for the Ascent Agena is designed and implemented to provide an automated, computer-controlled testing complex. The computer and RF console arrangement at the launch base is located in the RF room of the Launch Operations Building (LOB) as shown in Fig. 6-6.

6.4.1 RF Communication System

The radio frequency (RF) communication system provides the capabilities for processing vehicle telemetry and tracking data. It is capable of receiving the signals via either of the two SGLS links. There is capability for routing the signals to the patch panels for power and frequency monitoring. The SGLS downlink receiver can accept any preselected

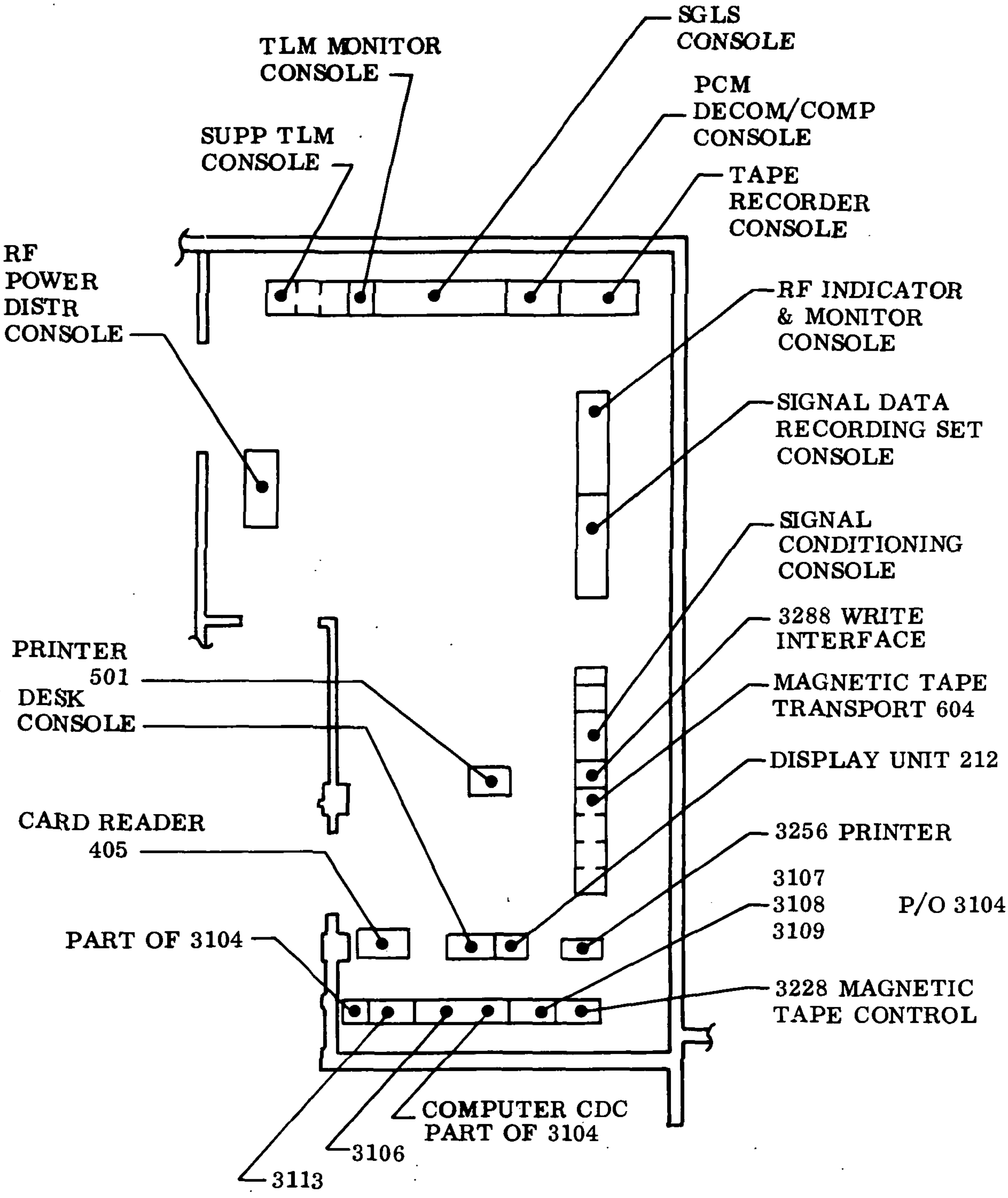


Fig. 6-6 RF Room Console Arrangement

phase modulated RF carrier in the frequency range from 2.2 to 2.3 GHz with the appropriate plug-in RF tuner.

6.4.2 Computer and Peripherals

The Control Data Corporation (CDC) GFE 3104 is a general-purpose computer with a 4,096 24-bit word core memory expanded to a 32,768 24-bit word core memory with a memory cycle of 1.75 microseconds. The computer and peripheral equipment for vehicle system test at the launch base is as follows:

Equipment	Quantity
Control Data Corporation (CDC) 3104, with two 3106 data channels, one 3107 data channel, and 32K of 24-bit words core	1
CDC 3101 with console typewriter	1
LMSC PCM decommutator/compressor	1
ELPAC 1003 PCM simulator	1
CDC 3288 write interface controller, including an interrupt expander	1
CDC 3256 printer controller/501 line printer	1
CDC 3228 magnetic tape controller/604 magnetic tape units	4
CDC 3291/221 cathode ray tube (CRT) display	1
CDC 3447/405 card reader	1

6.4.3 Launch Pad Hardline Checkout

The hardline checkout system consists of the following:

a. Electrical, auxiliary electrical, launch conductor, recorder, and ascent guidance system (AGS) consoles. These consoles, which are located in the Launch Operations Building, provide control and monitors for all umbilical functions, including the AGS (Fig. 6-7).

b. The signal processing console, which is located in the Launch Support Building (LSB), provides remote switching and signal conditioning for umbilical functions.

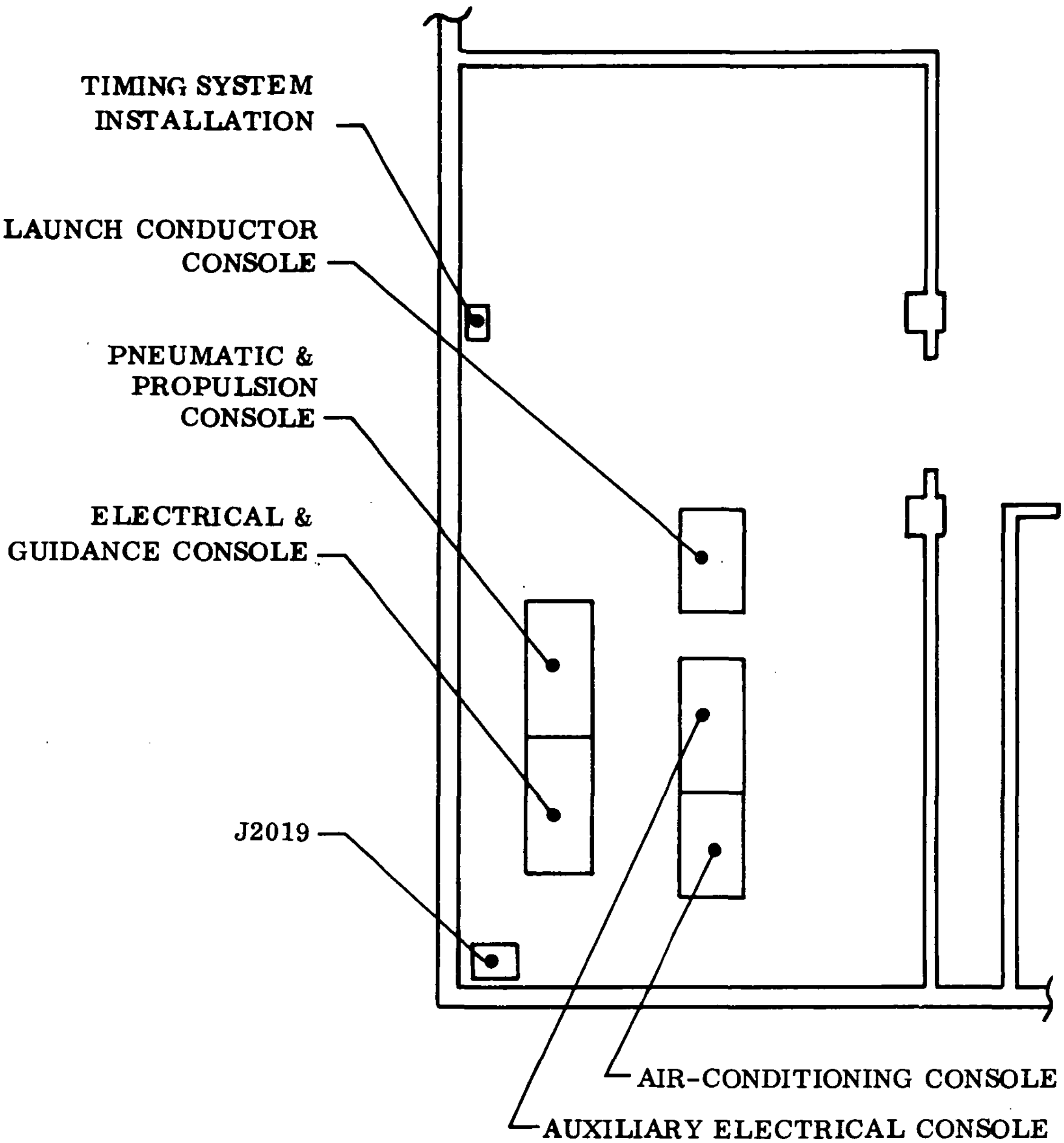


Fig. 6-7 Launch Control Room

c. Fast turnaround capability has been implemented through the use of dual sets of patchboards in the LOB and LSB interconnect consoles. All hardline cables terminate in their respective interconnect console, and it is at these central points reconfiguration between using programs is accomplished with preprogrammed patchboards.

The power control and monitor system consists of the electrical console, LOB and LSB power supply sets, the auxiliary power distribution console, and the umbilical junction box. This system provides and distributes all 28 vdc power used in the hardline system and also provides external and battery simulation power to the vehicle. The power system has an automatic voltage/current sensing system that will shut down all vehicle power if an out-of-tolerance condition exists on the vehicle bus.

The pad automated data evaluation system (PADE) accepts analog and discrete inputs from the hardline system, thereby providing a record on computer tape and printout of vehicle hardline commands and responses as a function of time, along with the corresponding vehicle telemetry record.

The launch complex timing system provides timing information based on time code word IRIG B (WTR Time). The countdown clock system provides visual display of countdown time in the LOB and is controlled from the launch conductor's console. Displays of WTR time are located in the control room and RF room of the LOB.

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